NASA CR71511

VOYAGER SPACECRAFT SYSTEM

FINAL TECHNICAL REPORT

TASK B

VOLUME B

DESIGN FOR THE OPERATIONAL SUPPORT EQUIPMENT

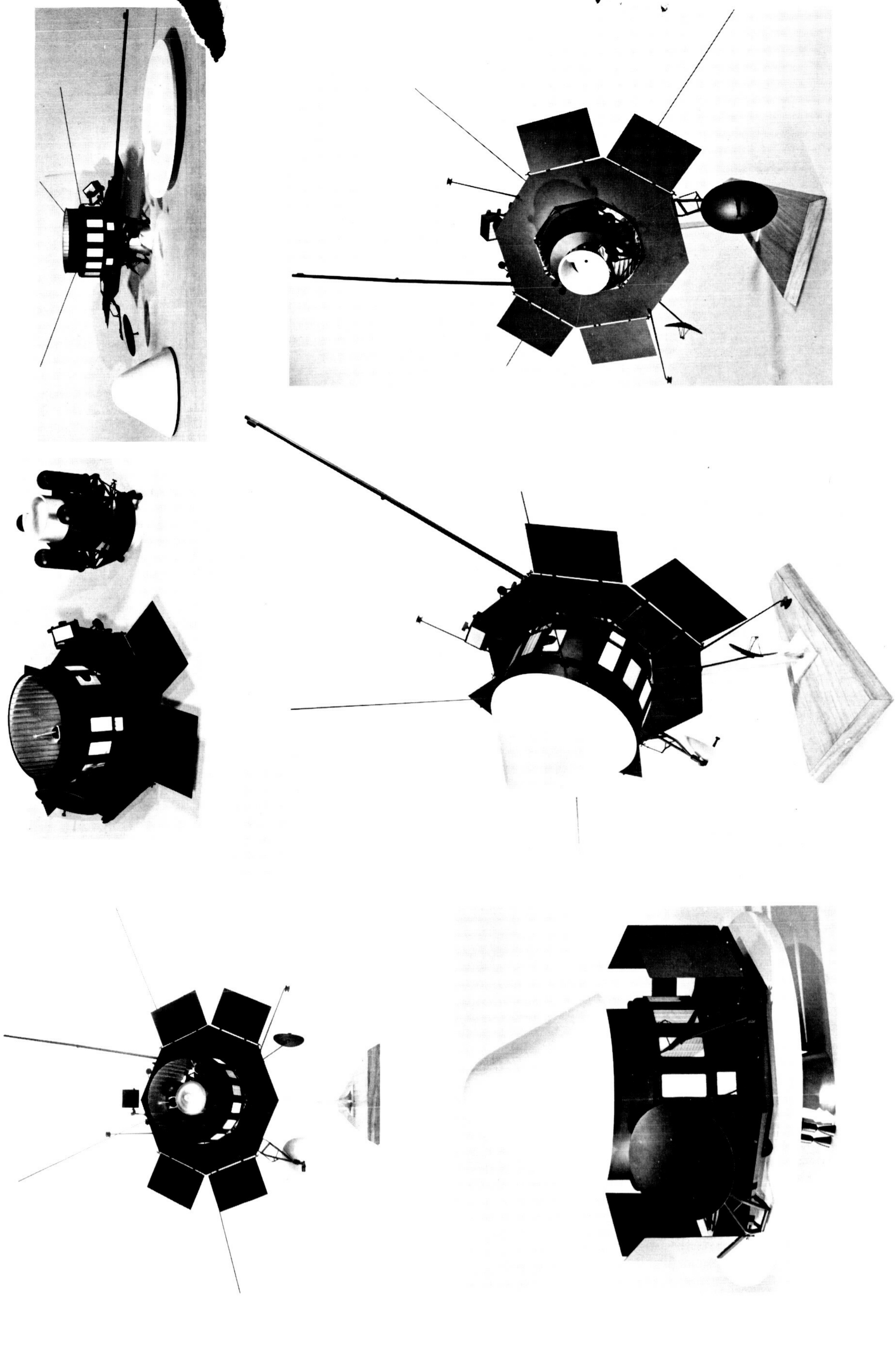
D2-82709-7

prepared for

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

UNDER
CONTRACT NO. 951111
JANUARY 1966
under NAS 7-100

THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON



Voyager Spacecraft 1/10 Scale Mockup

THE BOEING COMPANY
AEROSPACE GROUP • P. O. BOX 3707 • SEATTLE, WASHINGTON 98124

January 21, 1966

IN REPLY REFER TO

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California

Gentlemen:

The Boeing Company is pleased to submit the technical reports of the work accomplished under Voyager Phase 1A, Task B. Together with the reports of Task A, they represent to us a substantial contribution to our understanding of the objectives of the Voyager Project. As a corollary, it is believed they will demonstrate to you a dedication for, and a capability to perform, those tasks so important to fulfilling the Spacecraft Contractor's responsibilities.

The recently announced delay in the Voyager Program will test the dedication of all parties concerned. Despite our disappointment, we will not let this temporary setback deter our proceeding on a rational basis to be ready when funding levels again allow the program to proceed. It is important to note that the Task B documentation has been submitted as if no change had occurred in the Voyager Program. It should be recognized that corporate and group commitments contained in the documentation, in the areas of facilities and personnel, will be reconsidered when the Voyager program proceeds. At that time, Boeing will update and reaffirm the resources necessary to support the Voyager program.

Because of the cancellation of the Phase 1B, Part 2 Request for Proposal, we have chosen to highlight some of our management philosophy and organization rationale in a summary document, D2-82709-00. To place this in perspective, we have also summarized the salient features of the spacecraft design. Further, we have postulated some advanced missions, using the 1971 design, for further exploration of the solar system. This latter item is the basis for part of our continuing Voyager work.

Little more remains to be said except to restate that the Voyager Spacecraft System represents to us, more than a new product objective; it is an opportunity to participate in the extension of scientific knowledge in the universe and to contribute to national prestige.

Very truly yours,

THE BOEING COMPANY

Lysle A. Wood

Lysle A. Wood
Group Vice President-Aerospace

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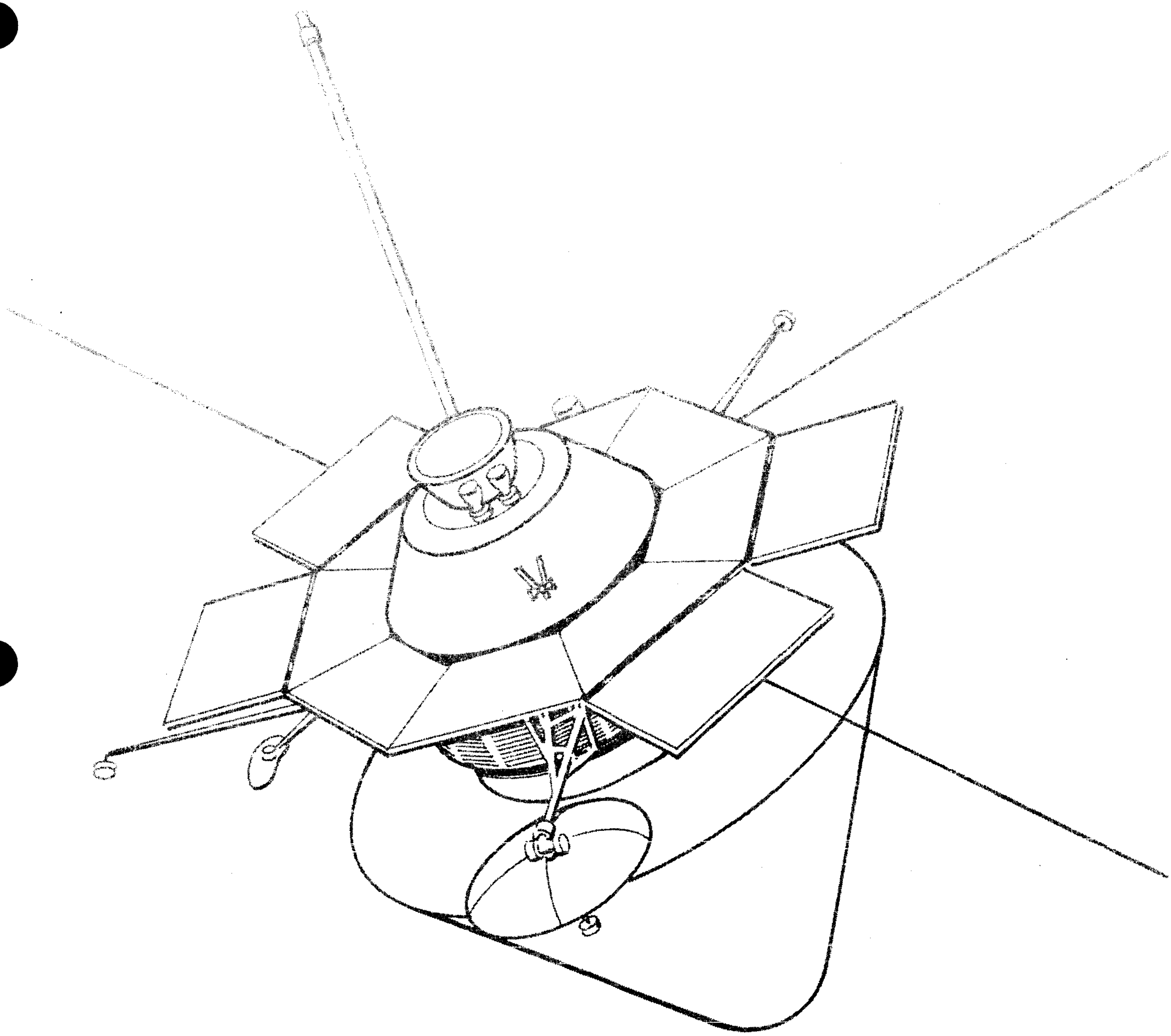


INTRODUCTION

This document, D2-82709-7 (Volume B), "Design for the Operational Support Equipment" is submitted by The Boeing Company in response to Contract 951111, Phase IA, Task B, dated November 2, 1965. It supplements D2-82709-3 submitted July 29, 1965, following completion of Phase IA, Task A. The complete technical report in response to Contract 951111, Phase IA, Task B consists of the following:

VOLUME A D2-82709-6 AND D2-82709-9*	PREFERRED DESIGN FOR FLIGHT SPACECRAFT AND HARDWARE SUBSYSTEMS PART I SECTION 1 - VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA SECTION 2 - DESIGN CHARACTERISTICS AND RESTRAINTS SECTION 3 - SYSTEM LEVEL FUNCTIONAL DESCRIPTION PART II SECTION 4 - FUNCTIONAL DESCRIPTION OF SPACECRAFT HARDWARE SUBSYSTEMS SECTION 5 - PROGRAM SCHEDULE AND IMPLEMENTATION PLAN
VOLUME B D2-82709-7	DESIGN FOR THE OPERATIONAL SUPPORT EQUIPMENT
VOLUME C D2-82709-8 AND D2-82709-10*	ALTERNATE DESIGNS CONSIDERED FOR SPACECRAFT PROPULSION SYSTEMS

*CLASSIFIED SUPPLEMENT TO VOLUME A AND C RESPECTIVELY

The highlights of the above documentation and management planning are summarized below.

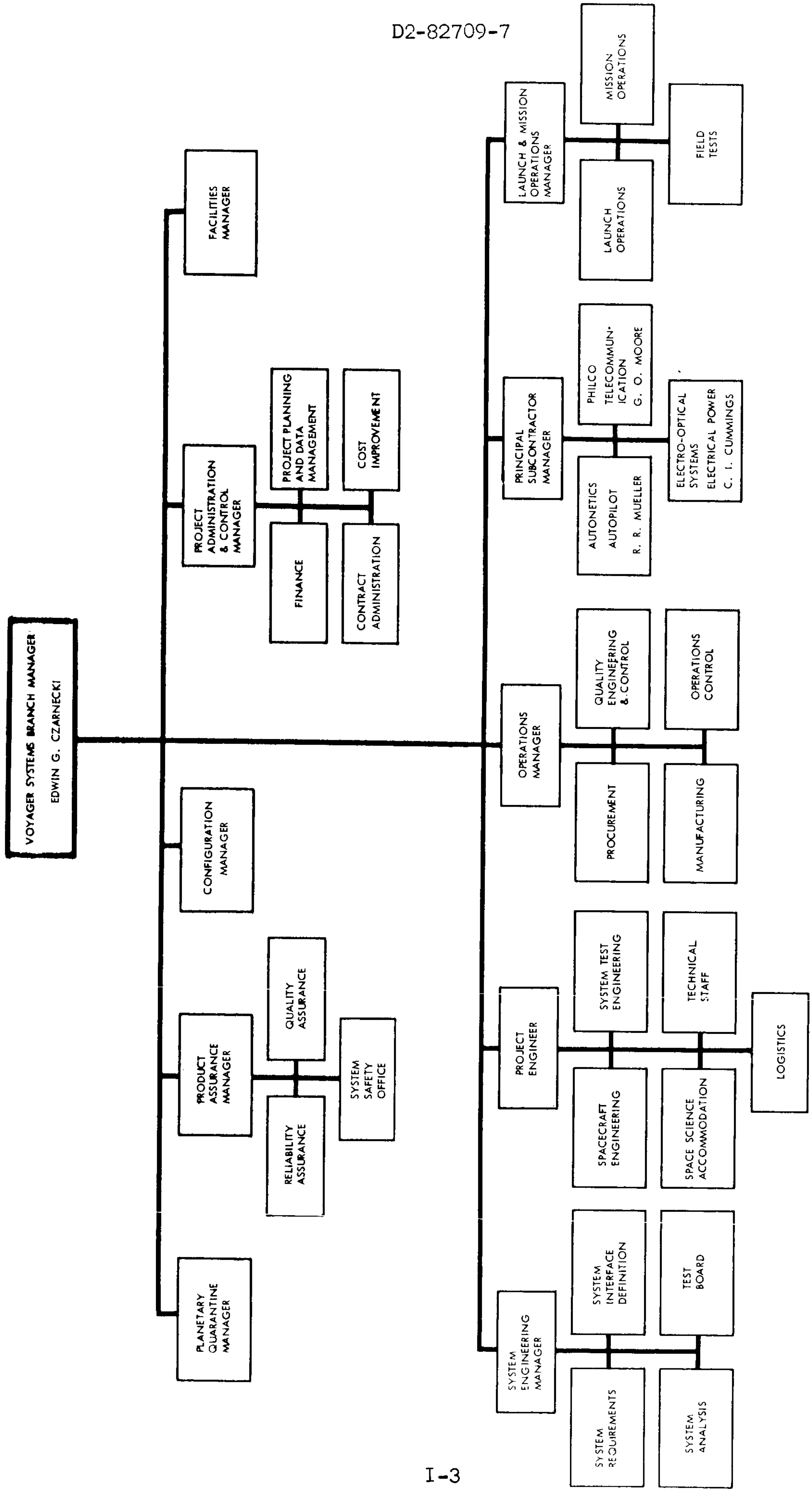
During the period covered by Contract 951111, Task B, Boeing has revised the preliminary design of the Voyager Spacecraft System in consonance with the statement of work. As part of this effort, Boeing has:

1) Verified and revised the requirements and constraints which are imposed upon the Voyager Spacecraft System by the Voyager 1971 Mission.
2) Reviewed and revised the preliminary Flight Spacecraft design for the Voyager 1971 mission, including the study of alternate designs for the spacecraft propulsion systems.
3) Selected a preferred design which reliably and effectively achieves the objectives of the 1971 mission.
4) Reviewed and revised the functional descriptions for the Flight Spacecraft and for each of its hardware subsystems.
5) Reviewed and revised the preliminary requirements and functional description for the Operational Support Equipment (OSE) necessary to accomplish the 1971 mission.
6) Updated and revised the schedule of the Voyager Implementation Plan.

The Boeing Voyager Spacecraft System organization, shown in Figure I-1, is under the direction of Mr. Edwin G. Czarnecki. Mr. Czarnecki is the single executive responsible to JPL and to Boeing management for the accomplishment of the Voyager Spacecraft Phase IA, Task B work and will direct subsequent phases of the program. He reports directly to Mr. George H. Stoner, Vice President and Space Division General Manager.

Although Boeing has capability in all aspects of the Voyager Program it is planned to extend this capability in depth through association with companies recognized as specialists in technologies critical to Voyager performance. The following team members have been chosen because of their experience and past performance:

D2-82709-7


VOYAGER SYSTEMS BRANCH MANAGER
EDWIN G. CZARNECKI
FACILITIES MANAGER
PROJECT ADMINISTRATION & CONTROL MANAGER
PROJECT PLANNING AND DATA MANAGEMENT
COST IMPROVEMENT
FINANCE
CONTRACT ADMINISTRATION
CONFIGURATION MANAGER
PRODUCT ASSURANCE MANAGER
QUALITY ASSURANCE
RELIABILITY ASSURANCE
SYSTEM SAFETY OFFICE
PLANETARY QUARANTINE MANAGER
LAUNCH & MISSION OPERATIONS MANAGER
MISSION OPERATIONS
LAUNCH OPERATIONS
FIELD TESTS
PRINCIPAL SUBCONTRACTOR MANAGER
PHILCO TELECOMMUNICATION G. O. MOORE
AUTONETICS AUTOPILOT R. R. MUELLER
ELECTRO-OPTICAL SYSTEMS ELECTRICAL POWER C. I. CUMMINGS
OPERATIONS MANAGER
QUALITY ENGINEERING & CONTROL
OPERATIONS CONTROL
PROCUREMENT
MANUFACTURING
PROJECT ENGINEER
SYSTEM TEST ENGINEERING
TECHNICAL STAFF
SPACECRAFT ENGINEERING
SPACE SCIENCE ACCOMMODATION
LOGISTICS
SYSTEM ENGINEERING MANAGER
SYSTEM INTERFACE DEFINITION
TEST BOARD
SYSTEM REQUIREMENTS
SYSTEM ANALYSIS

Autonetics, North American Aviation, Anaheim, California

Autopilot and Attitude Reference Subsystem
Mr. R. R. Mueller, Program Manager

Philco, Western Development Lab, Palo Alto, California

Telecommunication Subsystem
Mr. G. C. Moore, Program Manager

Electro-Optical Systems, Inc., Pasadena, California

Electrical Power Subsystem
Mr. C. I. Cummings, Program Manager

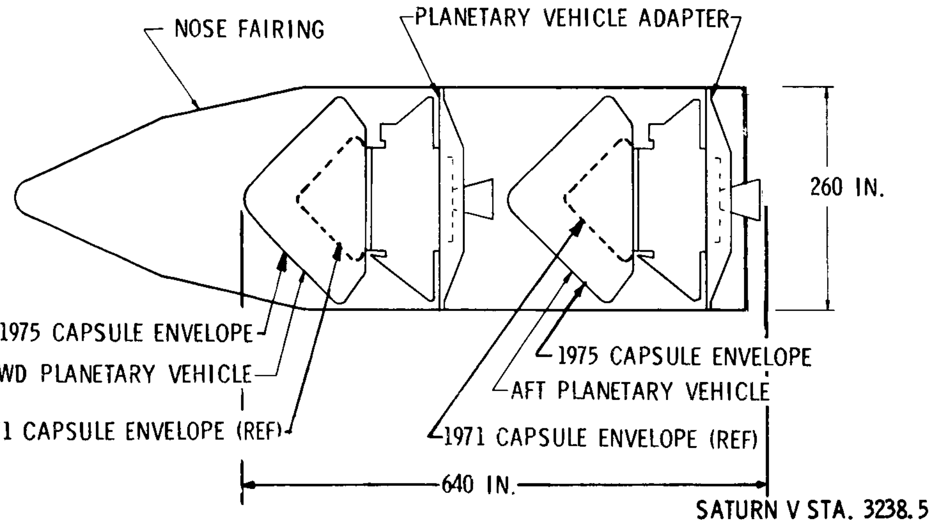
These subcontractor team members have been associated with Boeing on the Voyager Program for periods of 7 to 14 months. As a result of this, there has been sufficient exchange of information to make possible immediate implementation of the project with a Boeing team capable of satisfying the JPL requirements.

The preliminary design approach by the Boeing team has emphasized

- High probability of mission success.
- Conservatism, simplicity, selective redundancy in critical areas, and the use of Mariner experience.
- Versatility to accommodate a wide range of payload, mission, and data requirements.

The Voyager Flight Spacecraft, shown in Figure I-2, has the following principal features:

1) A capability to meet or exceed all mission requirements established in the Voyager 1971 Preliminary Mission Description.
2) A high probability (approximately 80 percent) of returning science data from at least one spacecraft in Mars orbit. The reliability

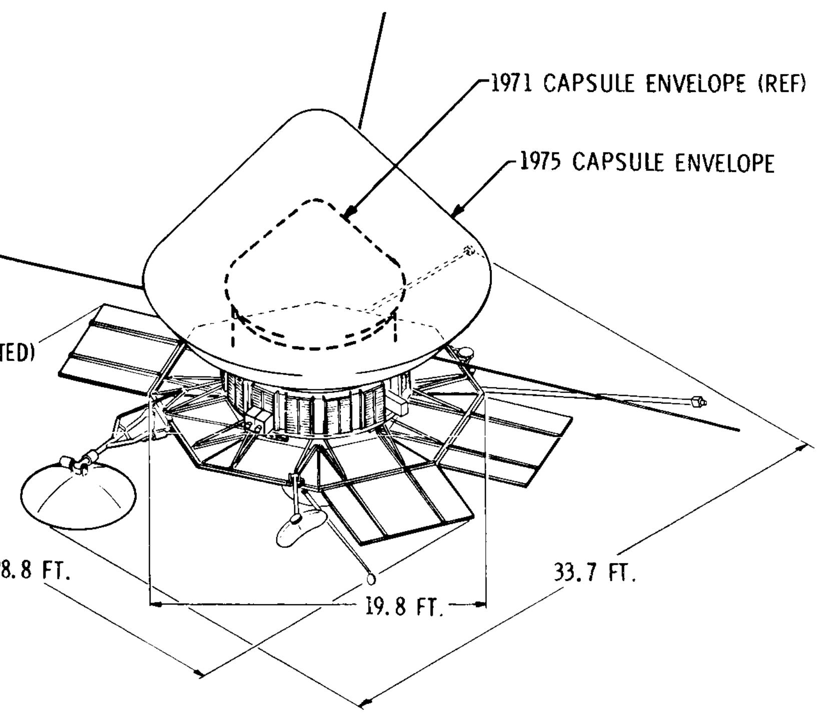


TOTAL SATURN V PAYLOAD ARRANGEMENT

CANDIDATE SPACECRAFT
SCIENCE PAYLOAD

SPACECRAFT BODY - MOUNTED INSTRUMENTS
- PLASMA PROBE
- COSMIC RAY TELESCOPE
- COSMIC DUST DETECTOR
- TRAPPED RADIATION DETECTOR
- ION CHAMBER
- MAGNETOMETER
- RF NOISE DETECTOR
- IONOSPHERE SOUNDER
- BISTATIC RADAR
- GAMMA RAY
- GRAVIMETER
- ULTRAVIOLET SPECTROMETER (SCAN MOUNTED)

SCAN PLATFORM - MOUNTED INSTRUMENTS
- INFRARED SPECTROMETER
- INFRARED SCANNER
- PHOTOIMAGING DEVICE



ASSEMBLED VIEW OF PLANETARY VEHICLE

EXPLODED VIEW OF VOYAGER FLIGHT SPACECRAFT & ADAPTER

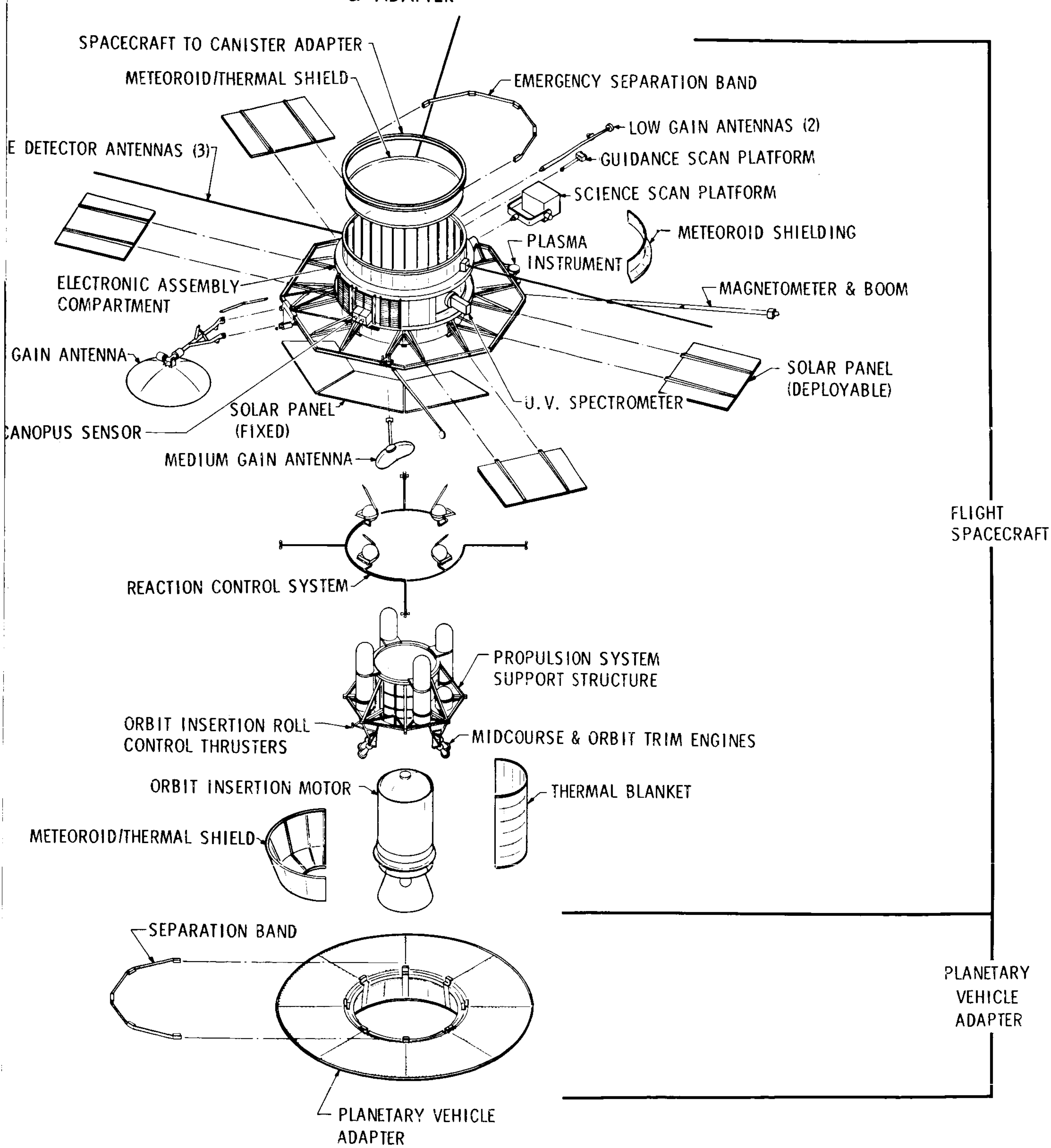


Figure I-2: Voyager Mars Mission Configuration of the Spacecraft Bus improved from .82 in Task A to .90 in Task B, primarily because of additional redundancy in the telecommunications system.

3) A spacecraft with subsystems sized to accommodate the range of anticipated Mars missions. The 1971 mission capability includes a 93-day launch period, periapsis altitudes as low as 400 km, orbit periods as low as 2.8 hours, and solar occultations as long as 3.7 hours.

4) A single propulsion module capable of fulfilling all Mars mission propulsion requirements from 1971 through 1977 without resizing or changing the propellant quantity.

5) Electrical and electronic systems designed so that no single failure will cause a catastrophic effect on the mission.

6) A computer and sequencer designed so that completion of a nominal mission can be accomplished with programs stored on-board and without ground command intervention unless required by trajectory dispersions or biasing. The ground system can override and back up these programs and command midcourse and orbit corrections when necessary.

7) Space is provided for 16 standard equipment assembly packages, 16" x 32" x 8.5", fastened to the 10-foot-diameter cylindrical structure and thermally interconnected. Fourteen of these are used in the preferred design, all of which employ standardized internal packaging. Thermal control of these assemblies is by space-facing plates radiating through Mariner C type bi-metallic-actuated louvers.

The Flight Spacecraft is 28 feet 10 inches wide, solar panel tip to solar panel tip. The height is 158" compared to a maximum allowable of 208 inches. The estimated weight is 1920 pounds for the Spacecraft Bus. A contingency of 180 pounds is therefore available for selective use during the detail design phase. The estimated weight of the propulsion module is 14,840 pounds with a contingency of 160 pounds (approximately 10 percent of the inert weights) available for use during the design phase.

Analyses and tradeoffs of the four specified Flight Spacecraft propulsion systems indicated that they were nearly equivalent in meeting the JPL specified requirements. The propulsion system selected is the modified Minuteman Wing VI second stage motor for orbit insertion and a hydrazine subsystem using four 200-pound thrust engines for trajectory corrections, and for orbit trim and vernier. The choice of this selected system was based primarily on the lower technical risk in the development of this system and the larger weight available for reallocation. In addition, it makes maximum use of available proven hardware.

A trade study was conducted between propulsion systems sized for 1971 - 1973 and for 1975 - 1977. The study showed that there were only minor differences and that a single design can be developed, tested and used without change for all missions, 1971 through 1977.

Wide variations in mission requirements are accommodated by the combined use of the solid motor augmented by the hydrazine system for orbit vernier. The performance of the selected propulsion unit exceeds all 1971 mission specification requirements. It provides an orbit insertion velocity increment in 1971 of 2.39 km/sec (2.2 km/sec design goal) with the 2000-pound capsule.

The hydrazine engines selected for trajectory and orbit correction maneuvers utilize a Shell 405 spontaneous catalyst. The engines are of the same type as those selected during the Task A preliminary design. They provide a total velocity change capability of 637 meters per second for the 1971 mission. The hydrazine subsystem has an engine-out capability without malfunction detection and switching. This is accomplished by canting the engines and using jet vane thrust vector controls to maintain the thrust vectors through the vehicle center of gravity. This, together with the use of proven components, results in a high confidence in the predicted reliability of 0.9960 for the preferred propulsion module.

The telecommunications subsystem is sized to meet the mission design requirements. It can accommodate higher data rates, and allow additional modes if such needs develop. The system selected uses a 50-watt traveling wave tube amplifier and a 6-1/2 foot diameter paraboloidal high-gain antenna with two axes of rotation. Complete coverage of the Earth is provided during Earth-to-Mars transit, orbit insertion, and orbiting flight. Space is available for growth to an 8 x 12 foot paraboloid. A maximum data rate of 7500 bps is provided with the $6\frac{1}{2}$ foot diameter antenna. The system has the potential for a data rate of 15,000 bps for a period of 20 days after encounter under worst case conditions. A 1260 bps backup mode is available during the first 100 days of Mars orbit. This is accomplished with a fixed Mariner C type paraboloidal antenna oriented to provide coverage of Earth during that period.

Five telemetry modes have been provided with data rates of 7500 bps for orbital use, 1260 bps for backup and late mission use, 80 bps for launch and interplanetary cruise, 1.64 bps for emergency use with the low-gain antenna, and an acquisition mode without data transmittal.

Data storage capacity is 3.8×10^8 bits in seven tape recorders. Recording and playback rates can be controlled redundantly through the Data Automation Equipment, Earth Command and the Computing and Sequencing Subsystem.

The Command Subsystem provides for two hundred (27-bit) stored and direct commands with growth provided for by expansion of the output combiner. Two complete, parallel command detectors and decoders with selection logic permits either detector to operate with either decoder to provide high reliability. The probability of executing a false command is several orders of magnitude less than the JPL requirement of 10^{-8}.

The Computing and Sequencing Subsystem controls the sequencing of time-dependent events during the Voyager mission. All functions for a nominal mission can be sequenced from launch through the end of orbital operations without command from mission control unless required by trajectory dispersions or biasing. The selected subsystem is a special-purpose programmable digital computer with an overall reliability of 0.986. It has a capacity for storage of 1024 (27-bit) words and a capability to execute 140 different commands. 700 words of storage are required to perform mission functions leaving a 30 percent reserve capacity in a standard size core memory assembly.

The Guidance and Control Subsystem is similar to that selected in Task A and draws heavily on Mariner and Ranger concepts. The Canopus and Sun sensors, the analog type autopilot, and the cold nitrogen reaction control system maintain cruise attitude within $\pm$ 0.3 degree. A planet sensor, limb detector, and terminator detector have been added to the Task A system. Single-axis ball and air bearing gyros and free rotor gas bearing gyros were re-examined. The free-rotor, gas-bearing Minuteman G6B gyro, modified to a higher torquing capability, was selected because of 1) demonstrated performance in the Minuteman application, and 2) a minimum number of units required for operational redundancy.

Reaction Control is by expulsion of cold nitrogen gas through coupled 0.125 pound pitch and yaw thrusters and coupled .035 pound roll thrusters. Sixteen separate thrusters are provided in a redundant configuration. Four titanium tanks contain 44 pounds of nitrogen. Under nominal conditions the nitrogen will last four years.

The Electrical Power Subsystem has been revised from the Task A design to satisfy new mission and physical constraints. Fixed and deployable panels were evaluated extensively. The selected solar panel array consists of 8 fixed trapezoidal panels (178 square feet), and 4 deployable rectangular panels (138 square feet) for a total of 316 square feet. This configuration meets power requirements for all mission periods and orbit selections, and in addition will meet major mission objectives if one panel fails to deploy. The solar electrical system provides 908 watts of gross power for spacecraft, capsule, and battery charging loads at the end of six months of orbital operation. The configuration can be tested in the Boeing Space Chamber with panels deployed.

Three silver cadmium batteries rated at a total of 2720 watt hours provide power for off-Sun periods up to 3.7 hours. Battery size and circuit design allow the mission to be completed if any one battery section fails. Prime power is distributed at 2400 cycles per second, single phase, 50 volts. Three sets of regulators, inverters and switching equipment are provided in a redundant configuration. This provides capability to operate all vehicle subsystems in event of a failure of any one power channel. Redundant 400 cycle per second inverters are provided for scan platform controls. Redundant precision oscillators are also provided.

The spacecraft structural arrangement is extensively revised from the Task A preferred configuration because of the larger and heavier propulsion module and increased capsule attachment diameter. Structural weight is 385 pounds and consists of 1) the primary structure assembly; 2) the external supports for appendages; 3) the capsule support and emergency separation assembly; and 4) an eight point Planetary Vehicle separation assembly. The primary structure is a 120-inch diameter magnesium shell, 85 inches long, of conventional semi-monocoque design. This shell provides direct support for attachment of 16 equipment modules (14 used) and for distribution of thermal loads between the assemblies. The Planetary Vehicle adapter is designed to support the spacecraft at eight points and provides uniform loading at the nose fairing interface.

The mechanisms employed for release, deployment and latching of deployed booms or linkages are the same as those proposed during Task A. Dual bridge-wire, pyrotechnic pin-pullers are used to release the pins holding the various components in their boost positions. Vinson-type actuators were selected for the deployment function; and spring-actuated, taper-pins are used to lock the components in their deployed positions. Self-aligning, spherical bearings are used for all hinge joints to counter any binding effects caused by thermal distortion; and sleeve bearings within the spherical bearings provide a second path of rotation, thus increasing the reliability of the system.

Four-segment, V-block separation bands are used to release the Planetary Vehicle from its adapter and also to effect emergency release of the Flight Capsule. Four pyrotechnic separation devices in each band assure a release reliability of .99992. Eight helical compression springs impart a total separation velocity of 1 foot per second.

The selected pyrotechnic subsystem follows the basic concept of the Mariner series in using capacitors and solid state switches. The pyrotechnic subsystem provides for a set of 21 command signals and 59 electro-explosive devices.

The Temperature Control Subsystem maintains the Spacecraft Bus, propulsion module and science instruments within specified operating temperatures throughout all the mission phases. The design approach, parts, and materials are similar to those used on Mariner "C". The equipment modules are coupled thermally and temperature control is accomplished by 52 square feet of bi-metal actuated louvers and high emittance radiator surfaces. The thermal dissipation capacity of the system is approximately 1200 watts, providing nearly 50 percent more capability than required to maintain gross thermal balance.

A science scan platform (GFE) is postulated to support the following science equipment: Infrared Spectrometer, Infrared Scanner, and two television cameras. This platform, with two-axis gimbal drive, provides the science instruments with clear views of Mars. An Ultraviolet Spectrometer is mounted on the spacecraft body with adequate scanning capability.

Substantial additional study and analysis has been made of ways to meet the planetary quarantine requirements and of the resulting Flight Spacecraft design constraints. New data made available or developed since the Task A report are:

1) The new Martian atmosphere which affects both probability of capture and heating rate of contaminated ejecta.
2) Microorganism IR emissivity which has been determined by Boeing to be approximately 0.2 instead of the previously estimated value of 0.5 to 1.0.
3) Increased microbial kill due to low ultraviolet attenuation in the Martian atmosphere.
4) Reduction by a factor of 10^4 in the meteoroid environment at Mars and associated reduction in the amount of contaminated material spalled off the orbiting spacecraft.
5) Tests run by Boeing which demonstrate with a high confidence that hydrazine is self-sterilizing.
6) Firings of solid engines by Boeing which indicate that the microorganisms do not survive the hot firing.

Based upon the above, the approaches taken in each hardware area for the selected design are:

1) Micrometeoroid Ejecta--No surface sterilization is provided for the spacecraft, but study and analysis should be continued. The higher ultraviolet kill and the lower micrometeoroid environment reduces the probability of contaminating the planet to 2.8×10^{-5}.
2) Reaction Control and Thrust Vector Control Ejecta, Midcourse and Orbit Trim Pressurant--Sterilize or decontaminate the nitrogen, Freon, and hardware internal surfaces. Study further to assess ultraviolet kill.
3) Midcourse and Orbit Trim Engine--No sterilization of the propellant or propellant hardware in this system is provided because of hydrazine's self-sterilizing characteristics. Tests in Phase IB are required to verify that microorganisms are not ejected from downstream hardware in Mars orbit.
4) Orbit Insertion Engine--Based upon UV kill and hot firing indications, this engine is not sterilized. Further analysis and hot test firings in Phase IB are required to confirm data prior to initiation of engine procurement.

The OSE selected is a modest extension of Mariner concepts. Subsystem test sets are used as the basic building blocks for the System Test Complex. The System Test Complex employs a Scientific Data Systems general purpose digital computer in a Central Data and Control System for automatic control of the subsystem test sets and central data analysis and display. The total design emphasizes minimum new development to enhance mission success and cost effectiveness.

Several existing test systems were considered for use in System Test Complex design and traded off against the preferred concept which is an updated version of that proposed by Boeing in the Phase IA Task A submittal. Systems considered include the Apollo Acceptance Checkout

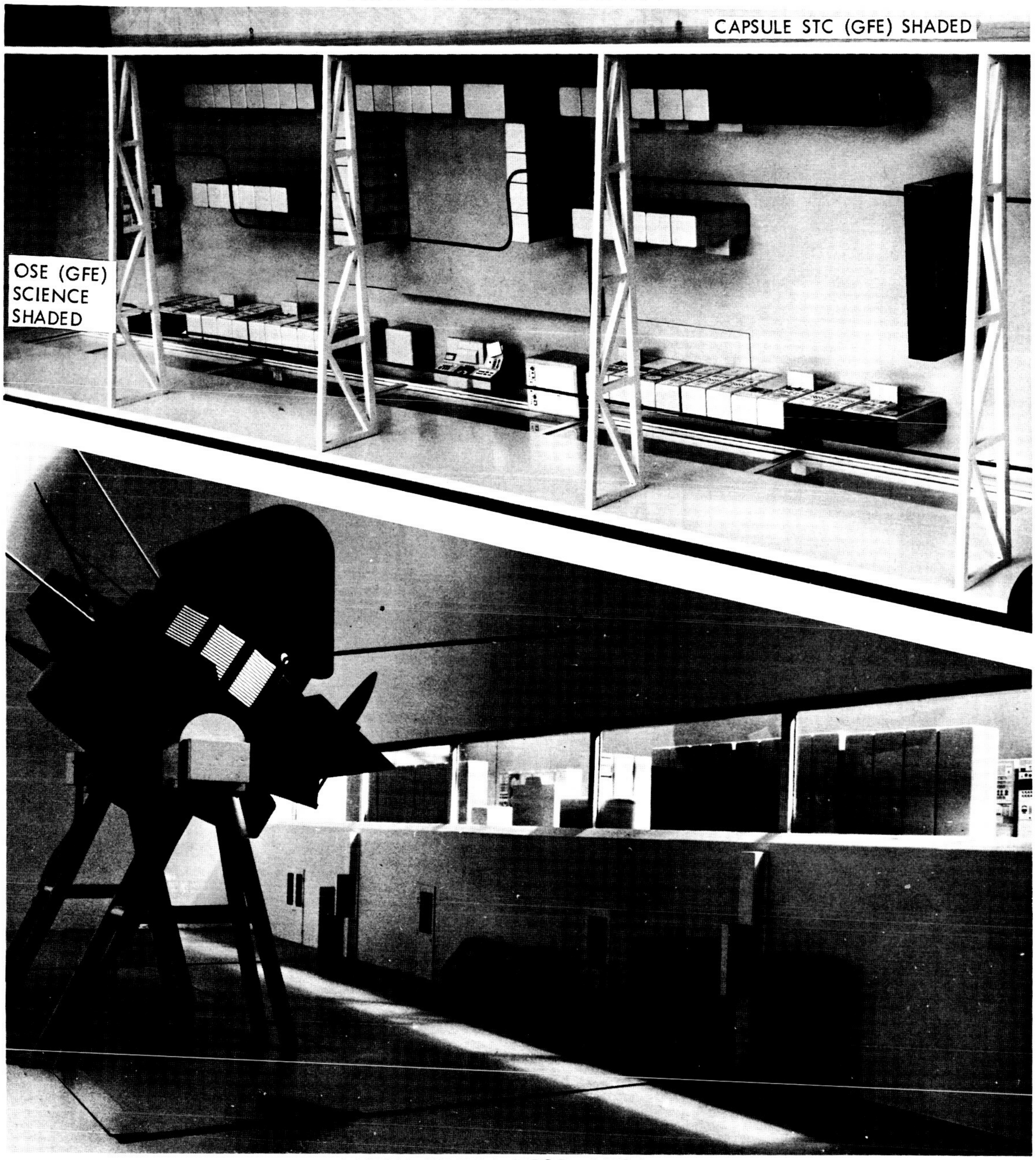
Equipment (ACE) and the Mariner C test equipment. The trade studies indicate that use of ACE would be either non-responsive to specification requirements or, if subsystem OSE were incorporated in the system, would be unnecessarily complex. Mariner C equipment does not include the required degree of central control and automaticity.

All requirements can be met with the preferred design which is well within current technology. It is planned that existing hardware be employed to a maximum degree in defining the Spacecraft System OSE and common components be employed wherever feasible.

The building block approach to design has also been applied to computer program development. Mission operations and test programs are assembled from sub-routines prepared in standard format in accordance with standardized software requirements. This minimizes software development time and costs and allows computer program preparation in parallel with equipment design.

Subsystem Test Sets are typically 1 to 9 standard racks containing equipment similar to that used in the Mariner Subsystem OSE. When elements of these are integrated with the SDS 920 (or 930) computer and appropriate interface adapters, they form a System Test Complex (STC) of approximately 55 cabinets (racks, output data units, and control consoles). Addition of the Mars surface lander capsule and Science Subsystem OSE brings the total Planetary Vehicle System Complex (STC) to about 76 cabinets. Figure I-3 shows a model of the STC, typical test facilities, and equipment. Elements of the STC are employed as an integral part of Launch Complex Equipment (LCE).

VIEW LOOKING DOWN



VIEW FROM TEST AREA

Figure I-3: System Test Complex And Equipment

A test team concept is planned wherein technical personnel experienced in spacecraft and OSE design, systems test operations, launch and mission operations, and spacecraft assembly and quality control will be formed into test groups. One of these teams will be assigned to each flight spacecraft and spare and will follow that vehicle from assembly through launch. Selected elements of the test team will continue to support mission operations for their spacecraft.

The Task B review and revision of the preliminary design for the Voyager Spacecraft System has emphasized conservative design, particularly in the use of proven equipment and techniques to the greatest extent consistent with system requirements. High reliability has been achieved through selection of space-proven components and through design of redundant capabilities into subsystems and equipment. The propulsion subsystem has been sized to achieve a range of flight trajectories and Mars orbits for missions in the years 1971 through 1977. The preferred Flight Spacecraft design provides mission versatility and capability for growth. As a result of the Task B activities, The Boeing Company has developed a design believed to be optimum for achieving objectives of the Voyager 1971 mission.

Conceptually, the OSE description which follows is similar to that of D2-82709-3, "Design for Operational Support Equipment" (Volume C), developed in July. No variation from "Voyager 1971 Preliminary Mission Description" or "Guidelines for the Voyager Spacecraft Contractor" was necessary in designing the Voyager OSE.

During the period since submission of the previous documentation, and in conformance with JPL's revised statement of work and associated specifications and guidelines, Boeing has performed the following OSE design functions:

1) Redeveloped the hardware and system functional flows;
2) Updated the requirements for the OSE and special test facilities necessary to accomplish the Voyager 1971 mission;
3) Updated the preliminary design for the Spacecraft OSE;
4) Updated the functional descriptions for the Spacecraft OSE;
5) Constructed a 1/20-scale mockup of the System Test Complex (STC) as envisioned in a typical spacecraft assembly/test situation to provide a visual concept of the relative space and arrangement requirements of the STC.

The engineering philosophy under which the OSE preliminary design work was done is one of maximum application of the Mariner C concept and the addition of automatic modes.

The major guidelines controlling the OSE engineering reported herein include:

1) Hardware Commonality--The OSE is designed to use common instrumentation and components throughout. For example the same digital voltmeters are used in the radio, pyrotechnics, central computer and sequencer, and power subsystem test sets.
2) Automation--The OSE at all test levels is designed for automatic test sequencing with provision for manual override. This feature provides the means for achieving a high degree of test repeatability and at the same time maintains the ability to do a single-step

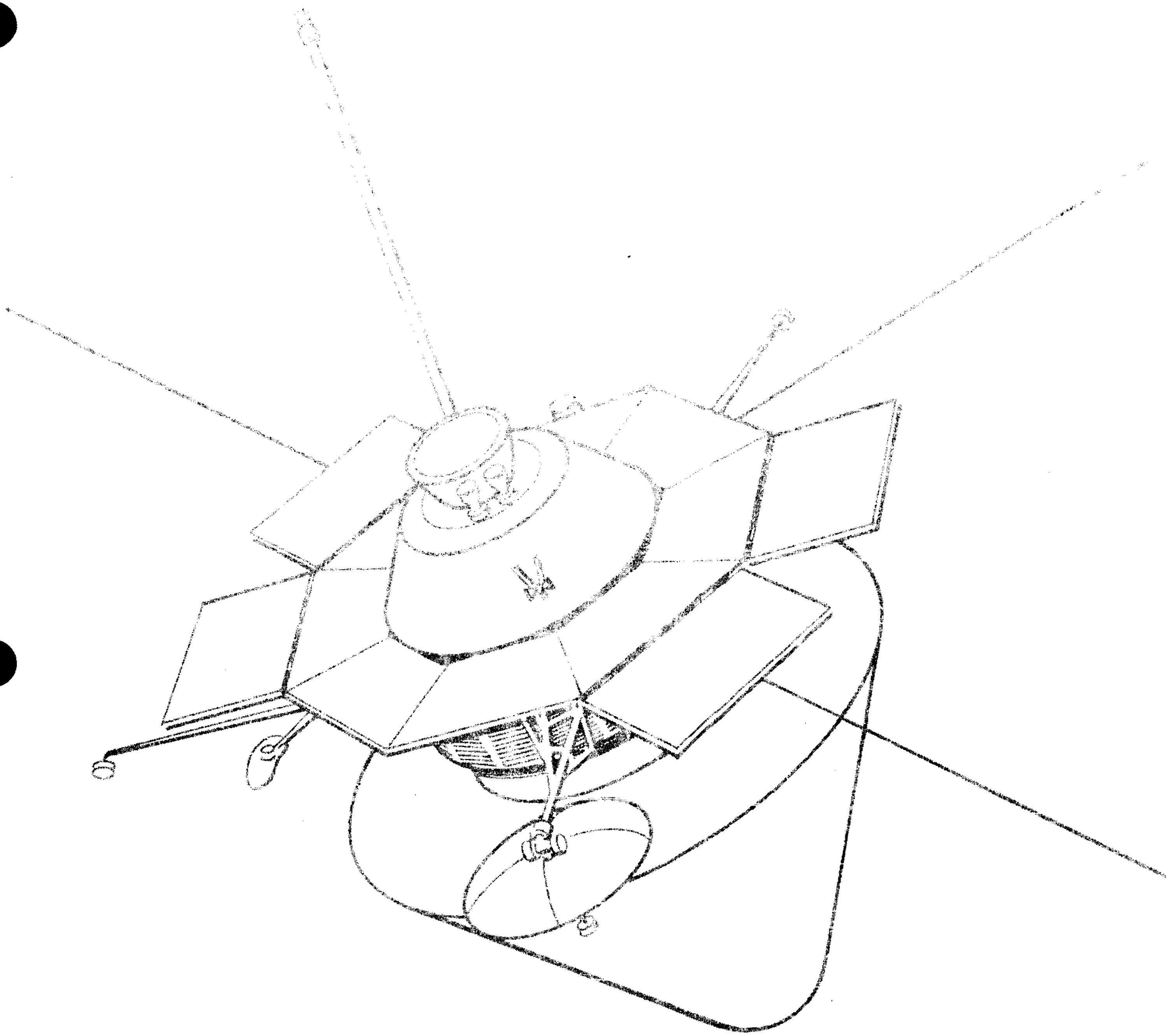
sequencing for both spacecraft and OSE hardware trouble-shooting and for fault isolation down to the replaceable-spares level.

3) Maintainability and Reliability--The previous two items, along with a self-check capability and critical-path redundancy, combine to provide a high reliability and high probability of launch readiness.

4) Transportability--The OSE, particularly that in the System Test Complex, is readily dismantled and may be transported by road or air.

5) Safety--The OSE is designed to present minimum hazard to either the mission hardware or operating personnel. Automatic monitoring and interlock circuitry minimize hazards to the mission equipment whether the OSE is in a normal or a malfunctioned condition.

6) Commonness--The basic Subsystem OSE (SS OSE) is used from subsystem type approval and acceptance through countdown and launch. The System Test Complex uses subsystem test equipment under the control of a central data system. The Launch Complex Equipment in turn uses the STC for prelaunch monitoring.



SUMMARY

SUMMARY

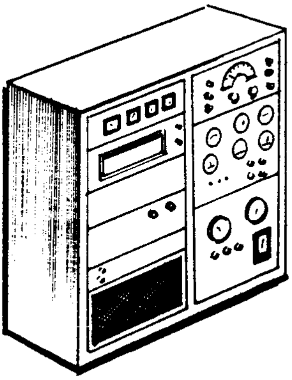
This document describes Boeing's concept and design of Operational Support Equipment (OSE) for the Voyager Flight Spacecraft (S/C). Spacecraft System OSE includes the spacecraft bus subsystem OSE (SS OSE), propulsion subsystem OSE, the government-furnished science subsystem OSE, System Test Complex equipment that integrates appropriate elements of SS OSE with a central data and control system, Launch Complex Equipment, Assembly, Handling and Shipping Equipment, Mission Dependent Equipment and Associated Software. This document also describes special test facility requirements. Figure S-1 provides examples to indicate the scope and variety of this equipment.

Individual Spacecraft subsystems are tested and checked each by its own subsystem OSE, which is designed, manufactured and first used by the same group that is responsible for the spacecraft subsystem hardware. These tests include type approval tests (TAT) and flight acceptance tests (FAT) at the completion of which the subsystem and the required subsystem operational support equipment (SS OSE) are delivered to the Flight Spacecraft assembly and test area. The subsystem-level OSE includes that equipment required for the Spacecraft Bus and its subsystems, for the Spacecraft Propulsion Subsystem, and the Science Subsystem. The latter item is government-furnished equipment to the Flight Spacecraft Contractor, and is defined in general terms to develop the interfaces with the STC and LCE.

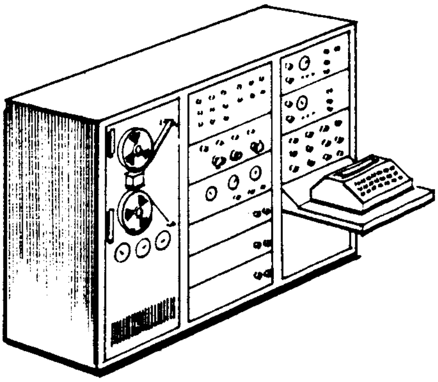
<u>System Test Complex</u>--The STC provides the system-level test capability required during assembly and testing up through Planetary Vehicle pre-

B L A N K

FLIGHT SPACECRAFT
(OSE)



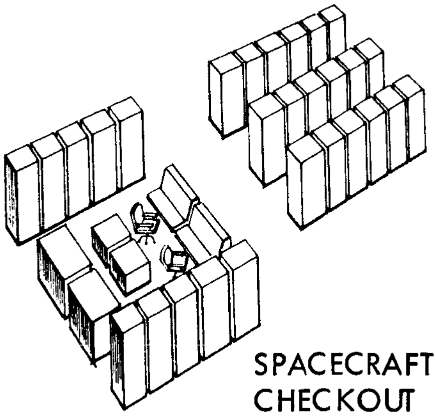
POWER SUBSYSTEM OSE



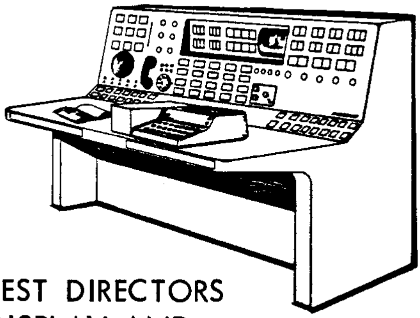
COMPUTING AND
SEQUENCING SUBSYSTEM OSE

GUIDANCE AND CONTROL
SUBSYSTEM OSE
RADIO SUBSYSTEM OSE
TELEMETRY SUBSYSTEM OSE
COMMAND SUBSYSTEM OSE
DATA STORAGE SUBSYSTEM OSE
STRUCTURAL AND MECHANICAL
SUBSYSTEM OSE
PYROTECHNICS SUBSYSTEM OSE
TEMPERATURE CONTROL
SUBSYSTEM OSE
PROPULSION SUBSYSTEM OSE
SCIENCE SUBSYSTEM OSE
INTERFACES
SOFTWARE

SYSTEM TEST COMPLEX (STC)



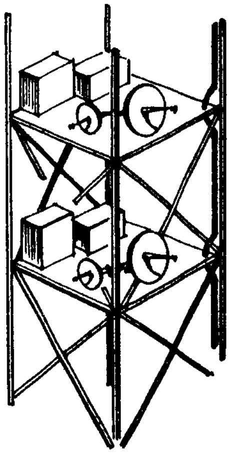
SPACECRAFT CHECKOUT FACILITY



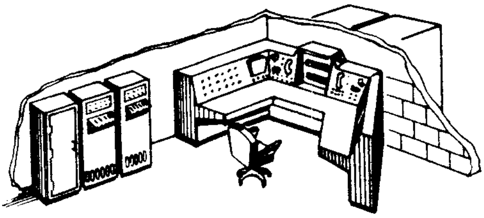
TEST DIRECTORS DISPLAY AND CONTROL CONSOLE

PARTIAL SUBSYSTEM OSE
CENTRAL DATA AND
  CONTROL SYSTEM
SPACECRAFT BUS
  SUBSYSTEM TEST ADAPTERS
SIMULATORS
  LAUNCH VEHICLE
  CAPSULE
  PROPULSION
  SPACECRAFT
POWER AND CABLING
ENVIRONMENTAL CONTROL
SOFTWARE

LAUNCH COMPLEX EQUIPMENT (LCE)



MOBILE LAUNCHER EQUIPMENT



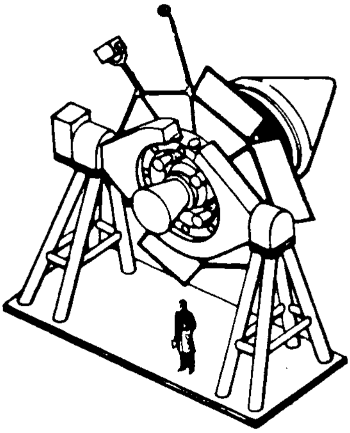
BLOCKHOUSE EQUIPMENT

PARTIAL STC(S)

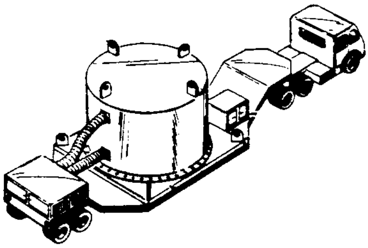
SCIENCE SUBSYSTEM
  MONITORS
PROPULSION SUBSYSTEM
  MONITORS
SIMULATORS
ENVIRONMENTAL CONTROL
POWER CONTROL
SOFTWARE

2

ASSEMBLY HANDLING AND SHIPPING EQUIPMENT (AHSE)



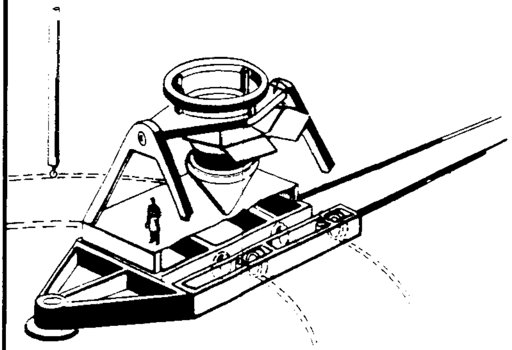
POSITIONER AND TEST STAND



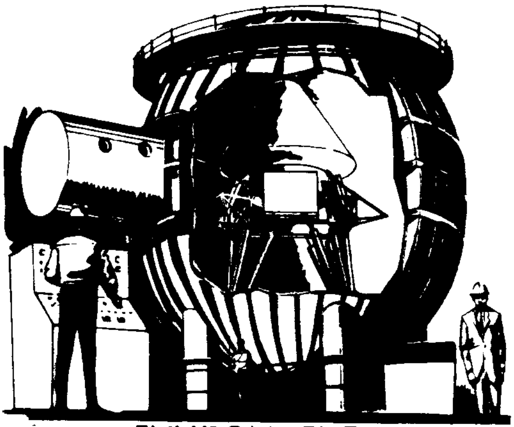
TRANSPORTER

ALIGNMENT FIXTURES
SAFETY EQUIPMENT
WORK PLATFORM SETS AND ACCESS EQUIPMENT
DOLLIES, SHOP TRUCKS, AND INSTALLATION DEVICES
PROTECTIVE COVERS
SHIPPING CONTAINERS
LIFTING DEVICES
SERVICING EQUIPMENT
INSTALLATION KITS AND ASSEMBLY JIGS
ENVIRONMENTAL CONTROL

SPECIAL TEST FACILITIES REQUIREMENTS



MAGNETIC MAPPING

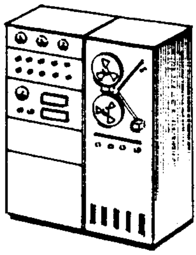


ENVIRONMENTAL TESTS

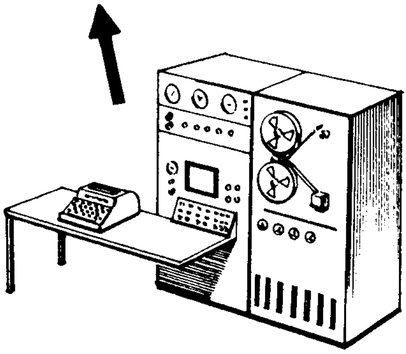
EXPLOSIVE SAFE AREA
POSTINJECTION TEST FACILITY
STERILIZATION FACILITY
SPACE SIMULATION FACILITY
ELECTROMAGNETIC TEST FACILITY
ORBIT INSERTION MOTOR TEST FACILITY
CELESTIAL SIMULATOR
LAUNCH ENVIRONMENT TEST FACILITY
ANTENNA RANGE

Figure S-1: OSE

SPACECRAFT MISSION
DEPENDENT EQUIPMENT (MDE)



DSIF
SUPPORT



SFOF
SUPPORT

MDE HARDWARE

MDE SOFTWARE

Concepts And Characteristics

flight checkout. The six major elements of the selected system are:

1) Appropriate portions of subsystem OSE (SS OSE).
2) STC/subsystem test set adapters.
3) A central data and control system (CDCS).
4) A voice communications system.
5) Simulators.
6) Test and checkout computer programs.

For STC use, the SS OSE sets are connected to the individual subsystems and to a group of test-set adapters. The adapters combine with the test sets to form interfaces between the STC central data and control system and the spacecraft subsystems. To properly sequence and interleave the various tests and to display and record results, an automated central data and control system, which includes a Scientific Data Systems 920 (or 930) Computer, is used to control the test process. Surveillance and supervision over this data system are exercised by the test conductor and his staff, who are supplied with the display, monitoring, and control facility required to interrupt or modify the sequence of computer routines that programs the accessing, processing, and distribution of test data. Figure S-2 shows the selected equipment.

The STC is supported by its own power, environmental control, and voice communications facilities, and operates in conjunction with the particular AHSE that holds and positions the spacecraft during its tests.

Simulators provide the interface characteristics of the Launch Vehicle, Flight Capsule, or Science Payload when required.

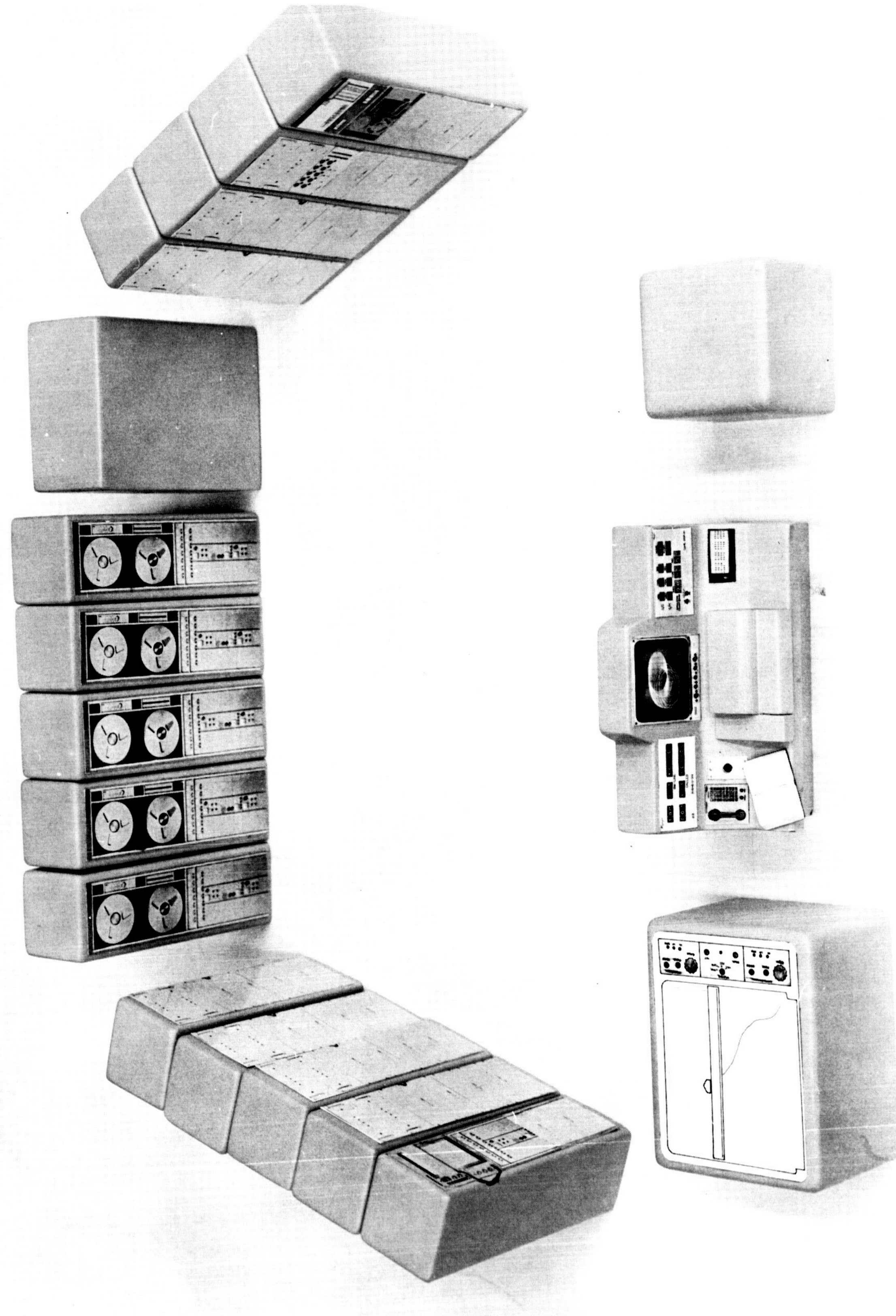


Figure S-2: Central Data And Control System

In arriving at a preferred STC concept, the following candidate systems were considered:

- Automatic test complex utilizing subsystem test sets. (Selected system).
- Updated Mariner C system.
- Modified - Acceptance Checkout Equipment, Apollo (ACE).
- Saturn S-IC electrical support equipment (ESE).
- Versatile automatic test equipment (VATE).
- Automatic checkout and readiness equipment (ACRE).
- Test equipment maintenance set (TEMS).
- Organization maintenance test station (OMTS).
- Drum programmed automatic tester (D-PAT).
- Multipurpose automatic inspection and diagnostic system (MAIDS)
- Computer controlled automatic checkout and evaluation equipment (SE-1000).
- OGO test equipment.

All candidate systems were evaluated against JPL stated requirements and constraints and also against derived requirements. This evaluation revealed that the three concepts, first listed above, were the most promising. Further analysis reduced these to the choice of the STC discussed herein because it satisfied the Voyager requirements with the least amount of equipment and new developments.

Launch Complex Equipment--The LCE determines the operational readiness of the Planetary Vehicle, monitors status during final countdown, controls the Planetary Vehicle during countdown, and conditions it for launch. The major elements of the LCE include the STC, blockhouse equipment, mobile launcher equipment, and terminal connector equipment. The LCE makes extensive use of the STC and incorporates the necessary signal conditioning, remote display, recording, and control equipment required because of geographic separation of LCE elements and personnel.

Since the Planetary Vehicles are enclosed in the nose fairing, data from the Planetary Vehicles are acquired through the rf link and umbilicals only; thus, fewer test data points are available to the LCE than to the STC. Flight commands may be sent via the rf link or the umbilical lines (or both), providing redundancy for effecting critical functions, and controlling and checking sequencing during countdown. The data flow through DSIF 71 to the STC facility or direct to the STC diplexer and are processed in the central data and control system (CDCS) and routed to the blockhouse Planetary Vehicle monitoring console. The Planetary Vehicle umbilical data pass through junction boxes, into signal conditioning equipment in the mobile launcher, through terminal connector junction boxes, and into the STC for processing and display. The data are also displayed at the blockhouse LCE monitoring and control center.

The data are processed, evaluated, and presented for display at the pertinent subsystem test consoles as well. These data, along with the status and response of the Planetary Vehicle subsystems to commands and stimuli, are time tagged and recorded. Selected status information is routed to the Planetary Vehicle monitor and control console for use by the test conductor.

In addition to the above, the LCE controls the environmental conditioning system, which provides cooled, dried, filtered air from a mobile air conditioning unit during explosive safe area (ESA) testing. After assembly, the cooled air is provided by the launch vehicle system. Voice communications between the launch pad, blockhouse, and launch control center are also provided as part of the LCE.

An emergency power system is incorporated in the LCE to ensure safe shutdown and conditioning of the Planetary Vehicle in the event of a facility power failure.

Mission-Dependent Equipment--The MDE is the hardware and software used in the Deep Space Network (DSN) that are unique to the Voyager Program. The hardware is required to buffer telemetry data into data processing systems, to provide for predetection recording of all telemetry data, and to insert command words into the DSIF modulator.

The software controls processing of command, telemetry data and scientific data at the DSIF and SFOF, and data processing for: 1) flight path analysis, 2) space science analysis, and 3) spacecraft performance analysis.

The MDE at the DSIF for handling command and telemetry data provides data and command optional modes for autonomous operation in the event of loss of communications to the SFOF or for test convenience. It also provides backup telemetry processing capabilities at the DSIF when required because of data-link failure or other SFOF equipment failures.

The MDE hardware and software are first required for use during system testing of the Spacecraft. They are developed concurrently with other major equipment items. Methods and techniques already in use by The Boeing Company for the NASA Lunar Orbiter Program will be used in the MDE production.

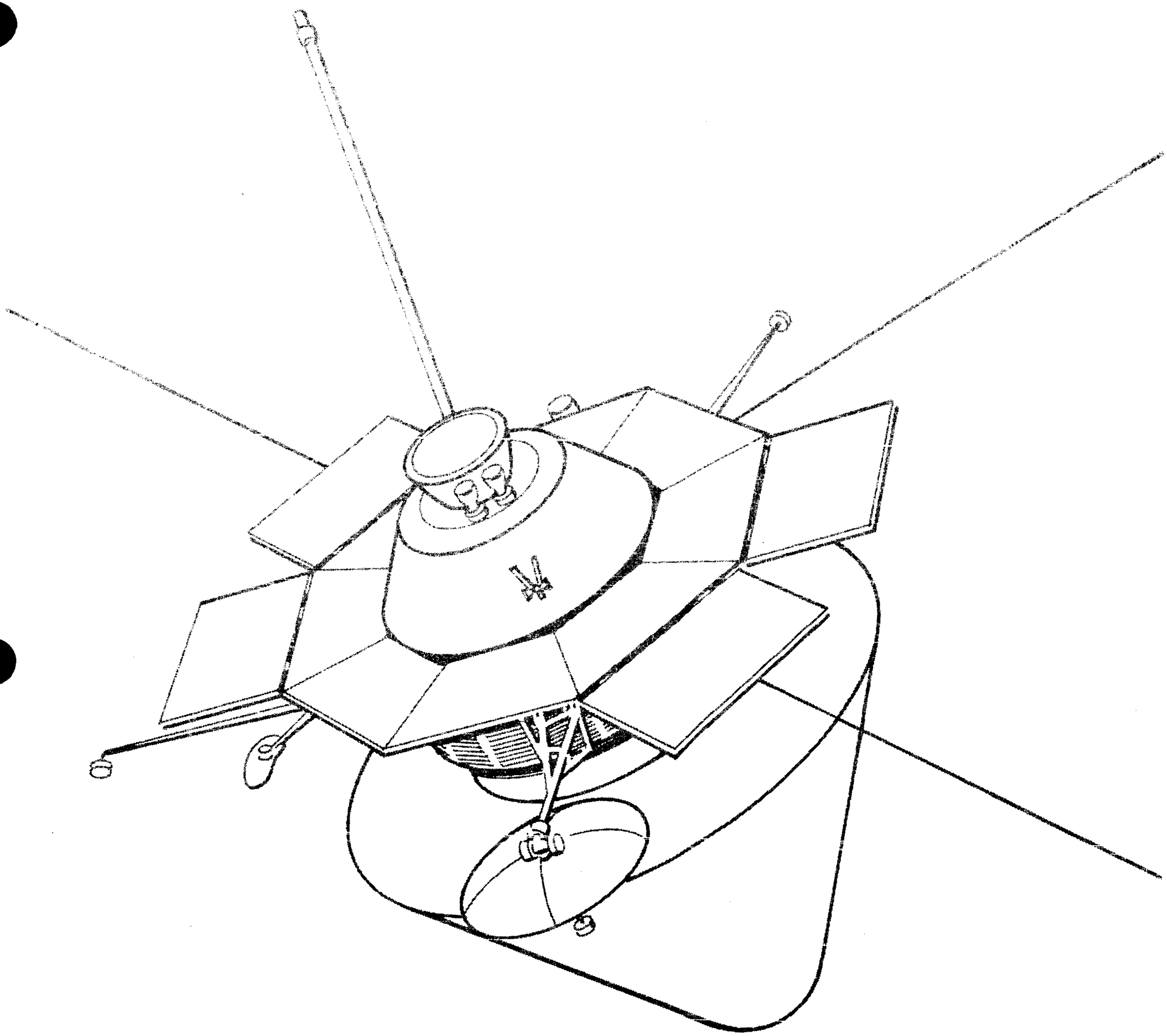
Assembly, Handling, and Shipping Equipment--AHSE includes hardware to lift, hold, and position spacecraft assemblies and OSE, and to maintain required environmental conditions. Approximately 130 items of AHSE have been identified.

The descriptions of the equipment include the functional requirements, interface requirements, performance characteristics, physical descriptions, reliability and safety considerations, and testing requirements.

In designing the AHSE, considerable emphasis has been placed on reliability, human engineering, materials selection, equipment and personnel safety, and maintainability.

Special Test Facilities--The requirements for the special test facilities were derived along with those of the AHSE and servicing requirements. Requirements for both the Seattle area and Kennedy Space Center are included.

Existing Boeing facilities, including a 30-foot diameter space chamber and solar simulator, together with the additional facilities planned will satisfy a major portion of Voyager requirements.



1.0 OBJECTIVES AND
DESIGN CRITERIA

1.0 OBJECTIVES AND DESIGN CRITERIA

The preliminary mission description dated October 15, 1965, states in part that "The primary objective of the VOYAGER PROGRAM is to carry out scientific investigation of the solar system by instrumented unmanned spacecraft..." Also, "The primary objective of the VOYAGER missions to Mars beginning in 1971 is to obtain information relevant to the existence and nature of extraterrestrial life..."

It follows, then, that development of the Flight Spacecraft Operational Support Equipment (OSE) includes the above objectives and, in addition, the following more explicit objectives.

1) Maximum effective support of the necessary subsystem- and system-level assembly, test, and operations phases by:
    a) Verifying the design and flight readiness of the Planetary Vehicle and its individual subsystems through controlled testing;
    b) Protecting and transporting the Planetary Vehicle through the various assembly, test, and launch facilities;
    c) Commanding the Planetary Vehicle during flight;
    d) Recovering the mission data via the Deep Space Network (DSN).
2) Maximum practical OSE reliability and operational availability. The mission-critical OSE reliability goal is 0.99 and is defined as the probability that the OSE will successfully perform its required function throughout the mission, starting with the terminal countdown and including thirty days of orbital operations. Successful performance is defined as:

a) Performance of the required functions without an OSE failure that causes a failure in the spacecraft;

b) Detection of all spacecraft and OSE failures that the OSE is designed to reveal;

c) Freedom from failures that would cause the mission to be re-scheduled once the terminal countdown is started.

3) Maximum use of design approaches that are logical extensions of Ranger and Mariner C OSE. For example, the Voyager OSE described herein is an extension of the Mariner C System Test Complex (STC) including an automated central data and control system and integration of the subsystem OSE.

4) Maximum use of off-the-shelf components and proven existing design concepts and equipment design within state-of-the-art limits anticipated for the 1971 launch period.

5) Maximum operational flexibility in areas where change can logically be expected. For example, the STC/Science-Subsystem OSE interface will likely experience several changes during the life of the program.

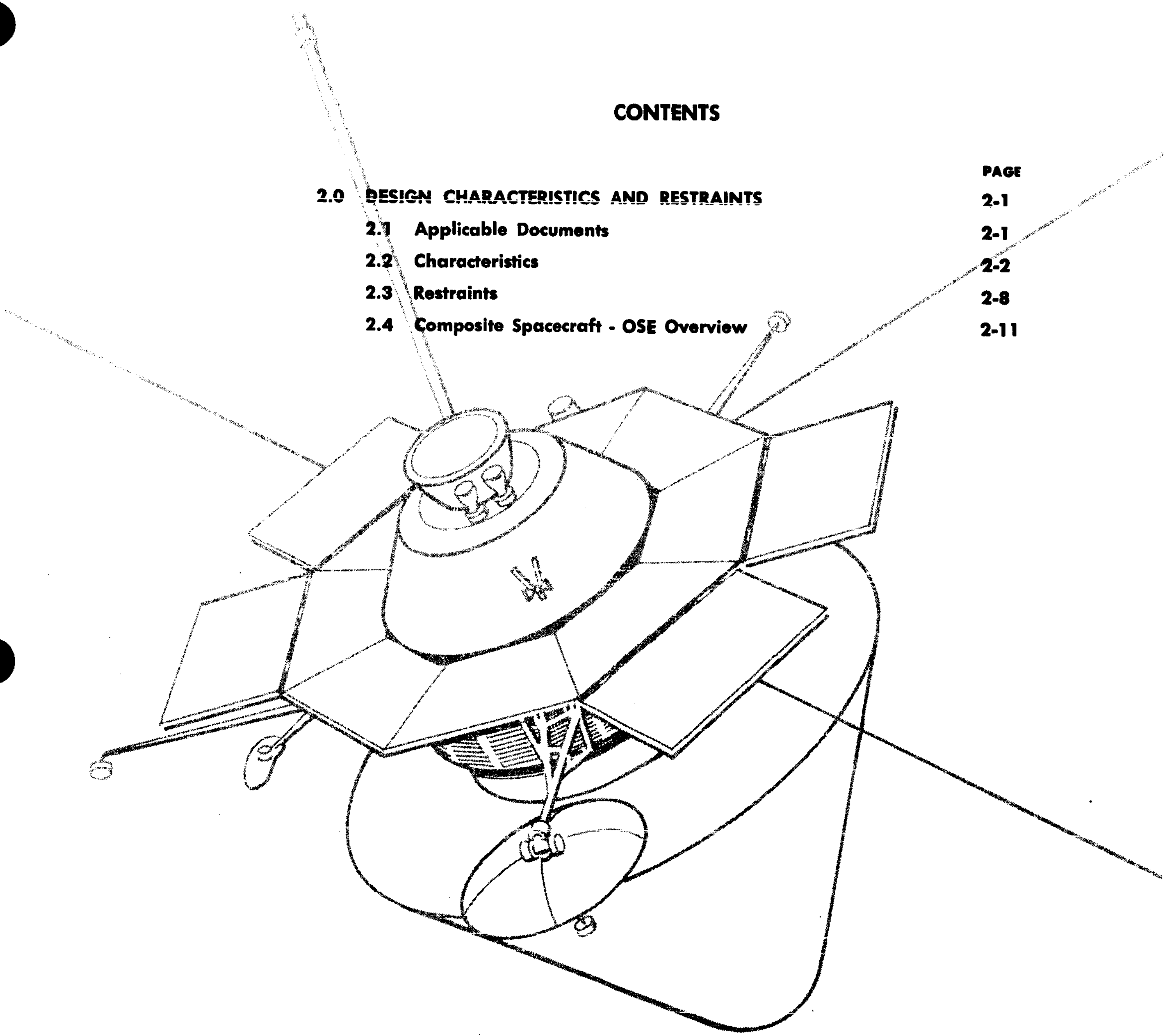
6) Adequate design for growth capability.

7) Proper balance of reliability and design for maintainability. Maintainability features necessary to meet the availability goal take into consideration equipment designed for ease of maintenance, preventive maintenance, and hardware commonality to lessen the logistics support burden on the system.

The objectives are goals that the design should support. It is necessary to have a means of determining how well the goals are being met, and, to this end, a number of standards or design criteria have been defined. Thus, the OSE major design criteria include the competing characteristics of the Voyager 1971 preliminary mission description and supplementary criteria as given below:

1) The degree to which the OSE design supports the overall mission probability of success;
2) The degree to which the OSE design facilitates performance of the mission objectives;
3) The efficiency and resulting cost economy;
4) The design and use flexibility to accommodate subsequent missions and added functions;
5) Simplicity and commonness of modules and submodules, which help simplify the design and minimize problems;
6) The effectiveness of the configuration in its support of the planned reliability program;
7) The ease of OSE operation and understanding of its functions;
8) Repeatability of operations and results;
9) Relative ability to use portions of the complex independently;
10) Relative freedom from coupled failure.

CONTENTS



2.0 DESIGN CHARACTERISTICS AND RESTRAINTS

The characteristics of the Operational Support Equipment (OSE) design and the restraints on the design are summarized in this section. Detailed design characteristics and restraints on the major elements of OSE can be found in Sections 3.0 and 4.0 of this document.

2.1 APPLICABLE DOCUMENTS

Below is a list of the source documents used as basic references for design reported in this volume. Other documents of importance to specific OSE systems are listed within individual system discussions (Sections 3.1 through 4.4 of this volume).

1) "Voyager 1971, Preliminary Mission Description," October 15, 1965.
2) Specimen Statement of Work, Phase IA, Task B.
3) MA-002-BB001-2A, "Guidelines for the Voyager Spacecraft Contractor," November 12, 1965.
4) "Preliminary Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification for," September 17, 1965.
5) "Voyager Environmental Predictions Document," September 17, 1965.
6) EPD-283, "Planned Capabilities of the DSN for Voyager," September 15, 1965.
7) 8907-A, "DSIF General Specification for Electronic Equipment," October 11, 1965.
8) MCO-50308-FAT, "Flight Acceptance Testing Specification."
9) MCO-50368-TAT, "Type Approval Testing Specification."
10) 3393-19-65, "General Specification, Telecommunications Design Control Table."

2.2 CHARACTERISTICS

The OSE designed to support the Voyager Spacecraft System consists of the following major elements: Spacecraft Bus Subsystem OSE (SS OSE), Spacecraft Propulsion Subsystem OSE, Science Subsystem OSE, System Test Complex (STC), Launch Complex Equipment (LCE), Assembly, Handling, and Shipping Equipment (AHSE), Mission-Dependent Equipment (MDE), operational software, and special test facilities (STF).

These elements are designed to provide a smooth transition from subsystem bench testing through evaluation of the spacecraft as it orbits Mars. This smooth transition is possible because of two fundamental characteristics.

First, the hardware is designed around a building block approach; for example, proven signal generators, in conjunction with such proven measurement equipment as digital voltmeters, are incorporated into a subsystem test set (SSTS), which, in turn, is mated to a subsystem to perform bench-level tests. At this point, a proven SSTS exists and is used as a component, or building block, in the next level of hardware--the STC. The STC employs, in its design, the proven SSTS and a central data and control system (CDCS), made up of a standard computer (SDS 920 or 930) and such other standard equipment as magnetic tape recorders. In turn, the LCE employs portions of the STC and also ties in the MDE.

The second characteristic is the unified software design. Starting with the bench-level tests, the software design has paralleled the hardware design to the extent that a standardized set of programs and procedures is used for generation of tapes used in the tape readers of the SSTS, generation of controls for STC and LCE, and for the data processing and trend analysis of SS OSE, STC, and LCE data.

As with the hardware, selected elements of this software are used as building blocks at the next level of testing. This basic characteristic of unified software design exists in all elements of the software shown below:

1) Mission Operations System/Deep Space Network (MOS/DSN) computer software;
2) LCE computer software;
3) STC computer software;
4) Test data reduction;
5) Trend analysis.

The two basic features of the OSE design (equipment commonality and building block approach) become more evident as the detailed characteristics of the hardware and software are discussed in subsequent pages.

Fundamentally, the SS OSE contains SSTS that can independently test and verify their respective spacecraft subsystems at any level of assembly and record all data necessary for analysis. Each SSTS can be manually controlled for investigative and diagnostic tests, and automatically controlled by tape programmer to allow repetitive testing or system-level tests. This feature of each SSTS--having its own programmer--minimizes the programming requirements at the next level of usage in the STC.

Each SSTS also has a standard interface within the System Test Complex that permits standard software programs to be used and enables standard self-check techniques to be employed at the STC. A standard interface also allows each subsystem to be optimized in its ability to test its subsystem without greatly influencing other elements of the STC.

The STC contains selected elements of subsystem OSE, (mainly the SSTS), selected elements of AHSE (mainly the two-axis test stand and miscellaneous handling equipment), selected elements of MDE (the telemeter and command processor), and a CDCS.

Section 3.1 of this document contains detailed information on the STC, the more significant features of which are:

1) Launch Vehicle, Flight Capsule, and Science Subsystem simulators to enable testing in absence of the Launch Vehicle, Flight Capsule, or the Science Subsystem;
2) A two-axis test stand that permits motion about the roll axis and a horizontal axis.
3) A central control console where the spacecraft test director can execute and monitor the system-level tests;
4) A CDCS that collects data in parallel with the SSTS, processes it into a standard format, and then records the data. Additionally, the CDCS processes and records the telemetry data.
5) A set of adapters for each subsystem test set provides a display of the processed telemetry data to the subsystem test engineers. These same adapters are used for displaying subsystem data during all levels of system tests.

These features allow the subsystem test engineer to correlate, in real-time, the data from his test set with data from the telemetry system (approximately 425 channels per S/C). Since these features are maintained throughout system-level tests, the transition to the launch operations phase is smooth and the same interface between man and machine is maintained by using the same building blocks in the next level of the OSE design--the LCE.

The LCE, described in detail in Section 3.2, employs elements of the STC equipment (mainly the adapters and the CDCS), elements of the MOS (mainly the DSIF 71 receiving equipment), elements of the launch vehicle system (mainly the umbilicals and mobile launch tower facilities), and some launch-complex-peculiar equipment (mainly the signal-conditioning equipment located on the mobile launch tower and the spacecraft coordinator's console located in the blockhouse). The salient features of this level of OSE design are:

1) A completely automatic monitoring and sequencing system that interfaces with the launch vehicle to maintain synchronization with countdowns and to shut down and safe the spacecraft automatically in the event of emergency;
2) A system that monitors and controls, simultaneously, two Planetary Vehicles enclosed in the fairing on top of the launch vehicle. The system also provides telemetry data and command links to the Capsule System OSE, the Science Subsystem OSE, the Spacecraft Bus OSE, and the Spacecraft Propulsion Subsystem OSE. This enables the cognizant engineers to maintain vigilance over their subsystem's performance;
3) Employment of DSIF 71 to receive and process the telemetry and to process and transmit commands to the spacecraft, permitting complete checkout and verification of these important data links and the associated operational software.

Involved with these data links is the spacecraft-peculiar MDE discussed in detail in Section 3.5.

The primary feature of the design is the buffering of telemetry and command data. The telemetry data require buffering after receipt at the DSIF or SFOF so that they can be processed by the DSIF and SFOF processing systems. Also, the commands coming from the DSIF or SFOF equipment must be buffered to provide the proper format for the transmitted command message to the spacecraft.

Software developed for Ranger, Mariner, and Lunar Orbiter provides the basis for Voyager computer software. However, additional software design is necessary. This design task is discussed in Section 3.6 and summarized here.

The major categories of MOS computer software support are (1) providing the mission-dependent elements of telemetry and command data handling software (TCD); (2) flight path analysis and command software (FPAC); (3) spacecraft performance analysis and command software (SPAC); and (4) mission integration and control software (MIC).

Some of the above computer software is complementary to other software; for example, the software used in part of the launch countdown (DSIF 71 telemetry and command links) is the same as the TCD used to support the mission. Another complementary feature is in SPAC, where computer programs for analyzing trend data from a spacecraft subsystem while the spacecraft is enroute to Mars are derived from the computer programs for trend analysis of Spacecraft System test data. Correlation of these analyses is enhanced by the commonality of software and the designs of the special test facility in which portions of the trend data are gathered.

Special test facilities are described in Section 3.4. The principal characteristics of these facilities are:

1) A simulation of the space environment to which the spacecraft will be subjected, the main features of which include a vacuum and thermal facility, a Sun simulator, and a simulated space heat sink;
2) A magnetic mapping facility for determining the effects of spacecraft residual magnetism on the magnetometer;
3) A launch environment simulator that closely matches the acoustic and vibration environment to which the spacecraft will be subjected;
4) A facility to determine the interaction of the Spacecraft Propulsion Subsystem and the attitude-control subsystem.

Each of the special test facilities is supplemented by special instrumentation and supported by selected portions of the AHSE. The AHSE is discussed in detail in Section 3.3.

AHSE is the category of equipment that includes the hardware necessary to lift, hold, position, service, control the environment of, and transport, Voyager assemblies and attendant OSE. The equipment is designed to minimize hazards to personnel and the spacecraft during hoisting, handling, testing, pressure-bottle filling, and rocket-propellant loading.

AHSE reliability is enhanced by use of designs and techniques proven on previous programs. For example, the Spacecraft Propulsion Subsystem draws heavily on Lunar Orbiter experience, the thrust-vector-control servicing unit is identical to one used on the HiBEX program, and the solar panel vans are similar to those used on the Mariner program.

The OSE design characteristics discussed above include human-engineering factors. Human-engineering factors are evident in the layout of the STC (Figure I-3). Workspace for OSE operator consoles has been investigated by Human Engineering for location, size, accessibility, and equipment configuration. More detailed investigation has been made reflecting such criteria as: body dimensions for the seated or standing operator, kick space at equipment racks requiring operator interface, writing surface dimensions, and visual access to the spacecraft at the STC. Preliminary considerations have also been given to possible psychological and physiological stress conditions (e.g., illumination, noise, and warning signals) that could constrain operator or director functions. Human-engineering design characteristics have been considered and implemented in all OSE preliminary design. At the STC, improved visual access to the spacecraft has been incorporated for all SSTS operators and the test director to provide additional spacecraft safety surveillance.

2.3 RESTRAINTS

Restraints imposed on the OSE design come from two primary sources. First are the restraints imposed by the statement of work and its enclosures, listed in Section 2.1. The most significant of these restraints are:

1) Launch opportunities and the launch window;
2) Use of Launch Complex 39;

3) Planetary quarantine requirements;

4) Experiments to be conducted;

5) Use of a single launch vehicle for two Planetary Vehicles.

The second source of restraints is the Voyager Spacecraft System analyses. These analyses were conducted principally under the precepts and disciplines of AFSCM 375-5 and NPC 500-X. The source data listed in Section 2.1 were reviewed, functional flows were developed, alternate approaches were defined and traded, requirements were allocated to design areas, alternate designs were developed, and the optimum design was selected. Of primary interest here are the restraints that resulted from these analyses. These restraints fall into two basic classes--functional and physical.

Functional--Functional restraints on OSE were derived from an analysis of the spacecraft processing. The major items of Seattle and Kennedy Space Center (KSC) processing, developed from the functional flows, are summarized in the following section. From these analyses, functional restraints were identified; detailed hardware restraints are shown in Sections 3.0 and 4.0 and general OSE restraints are as follows:

1) The STC must be transportable between test locations to maintain the mating of the spacecraft with its test equipment.

2) The special test facilities must provide multipurpose capability.

3) Packaging of the spacecraft for shipment to KSC must be compatible with the B-377PG-2 aircraft.

4) The explosive safe area (ESA) fueling facility must be capable of supplying hydrazine fuel, sterile nitrogen and freon.

5) The facilities at KSC must be capable of supporting the processing of three Planetary Vehicles in parallel.

6) The Spacecraft Propulsion thrust vector control and attitude control hardware must be decontaminated at KSC.

7) Assembly of the payload stack at the ESA requires a vertical clearance of 60 feet.

8) Communication circuits at KSC between facilities must be capable of handling the STC data rates.

Physical--Physical restraints on the OSE design are primarily derived from specific spacecraft handling, servicing, transporting, and testing requirements. The most significant of these physical restraints are as follows.

1) OSE elements used during spacecraft testing shall be designed for rapid assembly and disassembly and truck transportation between test facilities.

2) OSE designed for spacecraft transportation between test facilities shall not require spacecraft disassembly.

3) OSE designed to handle the spacecraft shall not magnetically interfere with or materially contaminate the spacecraft.

4) OSE used within the spacecraft testing environment shall be designed to prevent introduction to the spacecraft of contaminants from the OSE, and shall meet requirements of a Class 100,000 clean room.

5) OSE used in handling and in close proximity to the spacecraft shall be designed to permit thorough cleaning of all contact and exposed surfaces.

6) OSE shall be designed to allow standardization of functionally similar elements.

2.4 COMPOSITE SPACECRAFT-OSE OVERVIEW

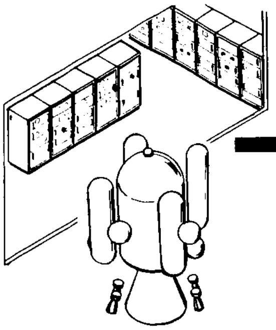
A spacecraft-OSE overview of Voyager processing in Seattle and at Cape Kennedy is presented in Figure 2.4-1. This figure is a summary of the more detailed flow diagrams and related information prepared by Boeing as a part of Task B activities. Shown are the general functions involved and the equipment and facilities required, with corresponding geographic locations, to provide overall visibility of the processing activity baseline used for the Task B study.

Seattle Processing--The concept is based on the science payload being delivered to the spacecraft contractor's plant for assembly into the spacecraft. After system-level acceptance tests of the spacecraft have been completed, Flight Capsules will be provided for combined system tests before the Planetary Vehicles are shipped to Cape Kennedy. The final tests at the factory duplicate, to the maximum extent feasible, the system interface tests to be conducted at Cape Kennedy. No modification or alteration of flight equipment other than repair of damage sustained in shipment, installation of ordnance items, decontamination as required, and the loading of propellants is planned after the flight hardware has left the factory.

Flight Spacecraft acceptance is shown after the Flight Spacecraft/Magnetic Mapping test and before introduction of the Flight Capsule to provide a convenient demarcation point between testing at the two major assembly levels. Planetary Vehicle acceptance occurs at the mission acceptance review.

B L A N K

AT SPACECRAFT CONTRACTOR

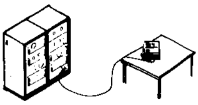


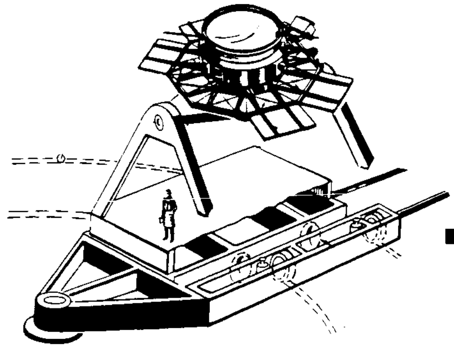
SCIENCE PAYLOAD

PRODUCE & ACCEPT S/C SUBSYSTEMS

Operational Support Equipment Used:
AHSE
SUBSYSTEM OSE

AT SUBCONTRACTORS





FLIGHT SPACECRAFT
DD-250 BUY-OFF

PROVIDE FLIGHT CAPSULE

PERFORM S/C MAGNETIC MAPPING

Operational Support Equipment Used:
AHSE
STC

KENT MAGNETIC MAPPING FACILITY

PE
AS
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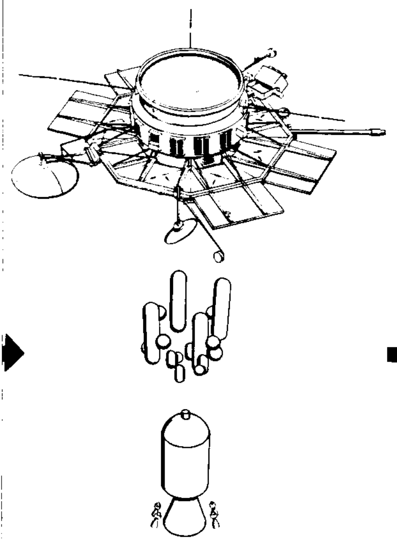
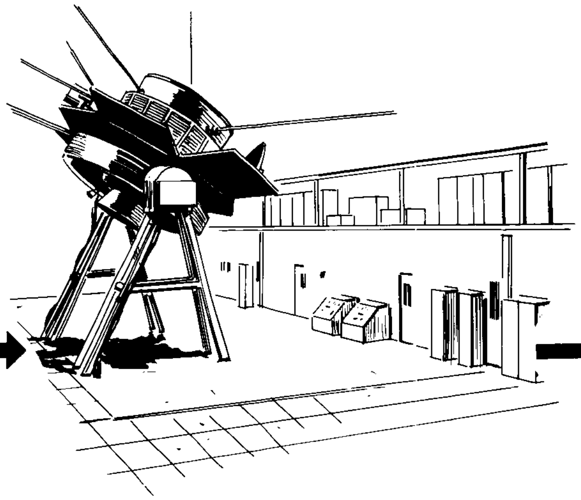


ASSEMBLE FLIGHT SPACECRAFT

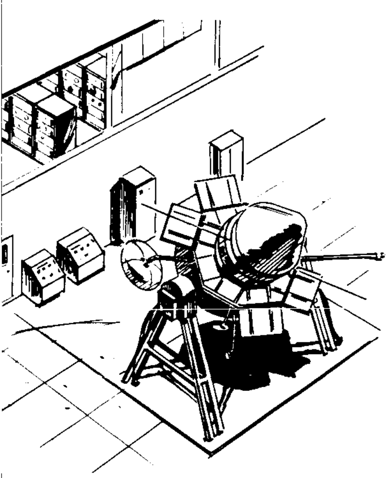
KENT ASSEMBLY & TEST FACILITY



STC ACCEPTANCE

Operational Support Equipment Used:
STC
S/C SIMULATOR

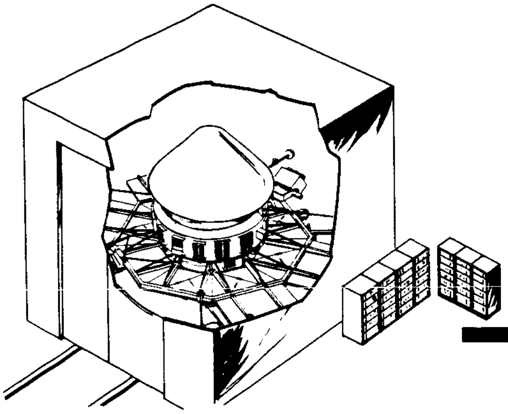
KENT ASSEMBLY & TEST FACILITY



RFORM RELAY LINK TEST
SEMBLE F/C TO FLIGHT S/C
EMETRY CHANNEL CALIBRATION
RFORM P.V. SYSTEM TEST

erational Support Equipment Used:
AHSE
STC

NT ASSEMBLY & TEST FACILITY

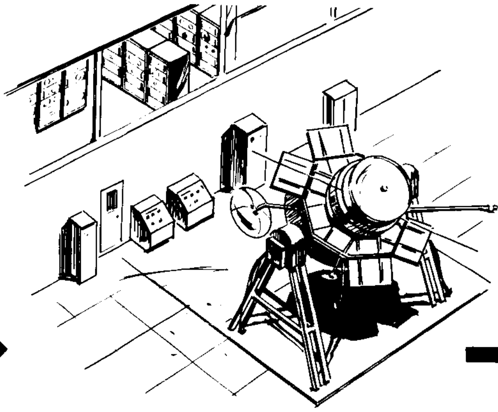


PERFORM P.V. EMI TEST

Operational Support Equipment Used:
AHSE
STC

KENT ASSEMBLY & TEST FACILITY

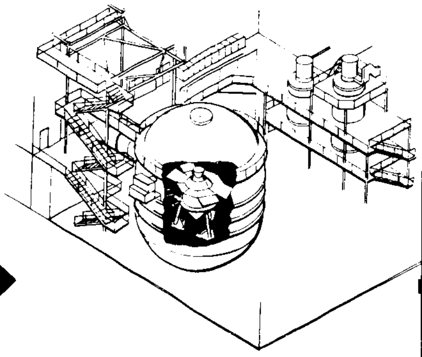
2



SPACECRAFT GROUND INTEGRITY TEST
INITIAL POWER APPLICATION TEST
SUBSYSTEM TEST
INTERSUBSYTEM TEST
TELEMETRY CALIBRATION TEST
SYSTEM TEST
PARAMETER VARIATION TEST

Operational Support Equipment Used:
AHSE
STC

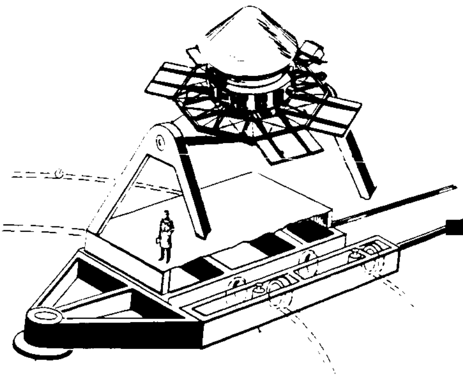
KENT ASSEMBLY & TEST FACILITY



FREE MODE TEST
SPACE SIMULATION TEST
THERMAL
VACUUM

Operational Support Equipment Used:
AHSE
STC
LCE

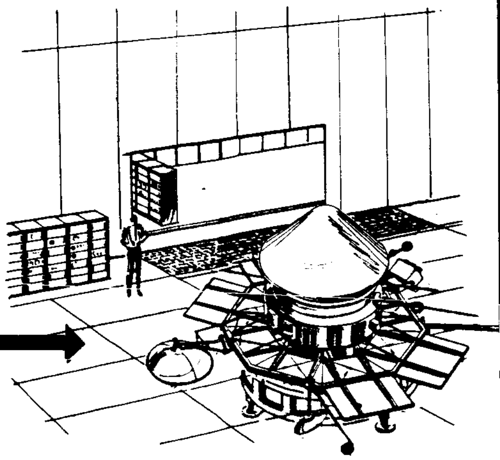
KENT-VACUUM CHAMBER FACILITY



PERFORM P.V. MAGNETIC MAPPING

Operational Support Equipment Used:
AHSE
STC

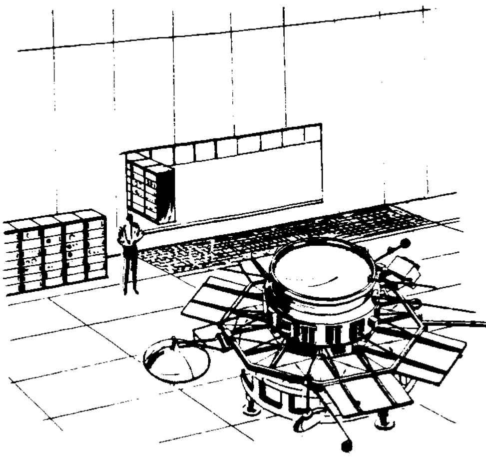
KENT MAGNETIC MAPPING FACILITY



DETERMINE P.V. WEIGHT & C.G.

Operational Support Equipment Used:
AHSE

KENT ASSEMBLY & TEST FACILITY

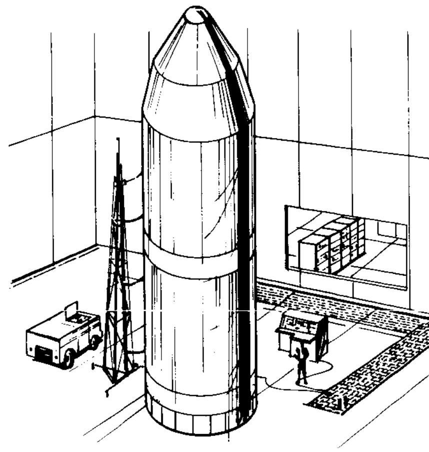


WEIGHT & C.G. MEASUREMENT

Operational Support Equipment Used:
AHSE

KENT ASSEMBLY & TEST FACILITY

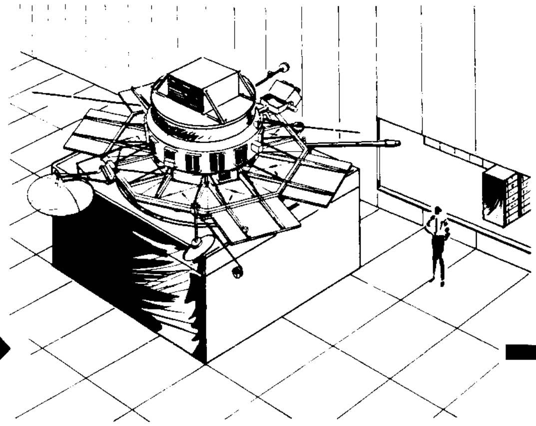
PROVIDE FLIGHT SHROUD ELEMENTS AND UPPER NOSE FAIRING



ASSEMBLE PAYLOAD
PERFORM COOLING TEST
PERFORM S/C-LAUNCH VEHICLE INTERFACE TES
S/C-MDE INTERFACE TEST
PERFORM DUMMY RUN COUNTDOWN
PERFORM SIMULATED LAUNCH
EMI TEST

Operational Support Equipment Used:
AHSE
STC
LCE
MDE

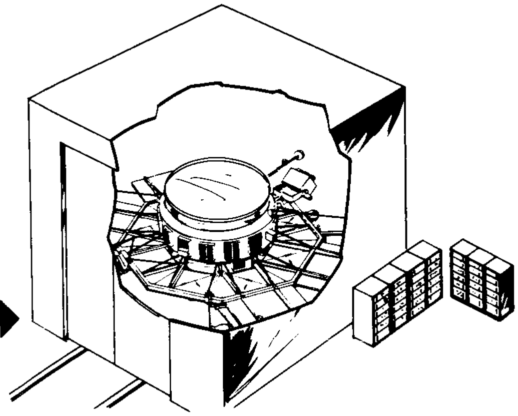
KENT ASSEMBLY & TEST FACILITY



PERFORM VIBRATION TEST

Operational Support Equipment Used:
AHSE
STC

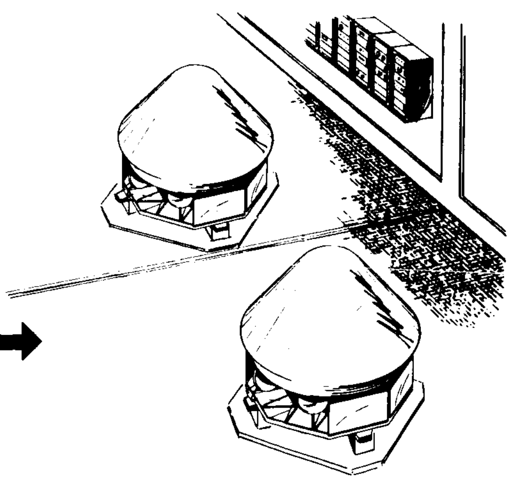
KENT ASSEMBLY & TEST FACILITY



PERFORM S/C EMI TEST

Operations Support Equipment Used:
AHSE
STC

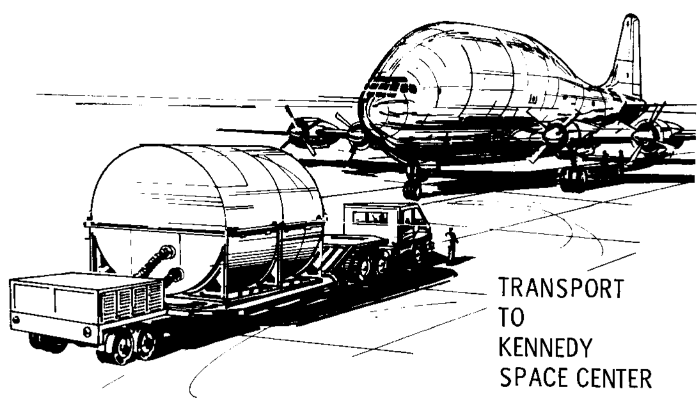
KENT ASSEMBLY & TEST FACILITY



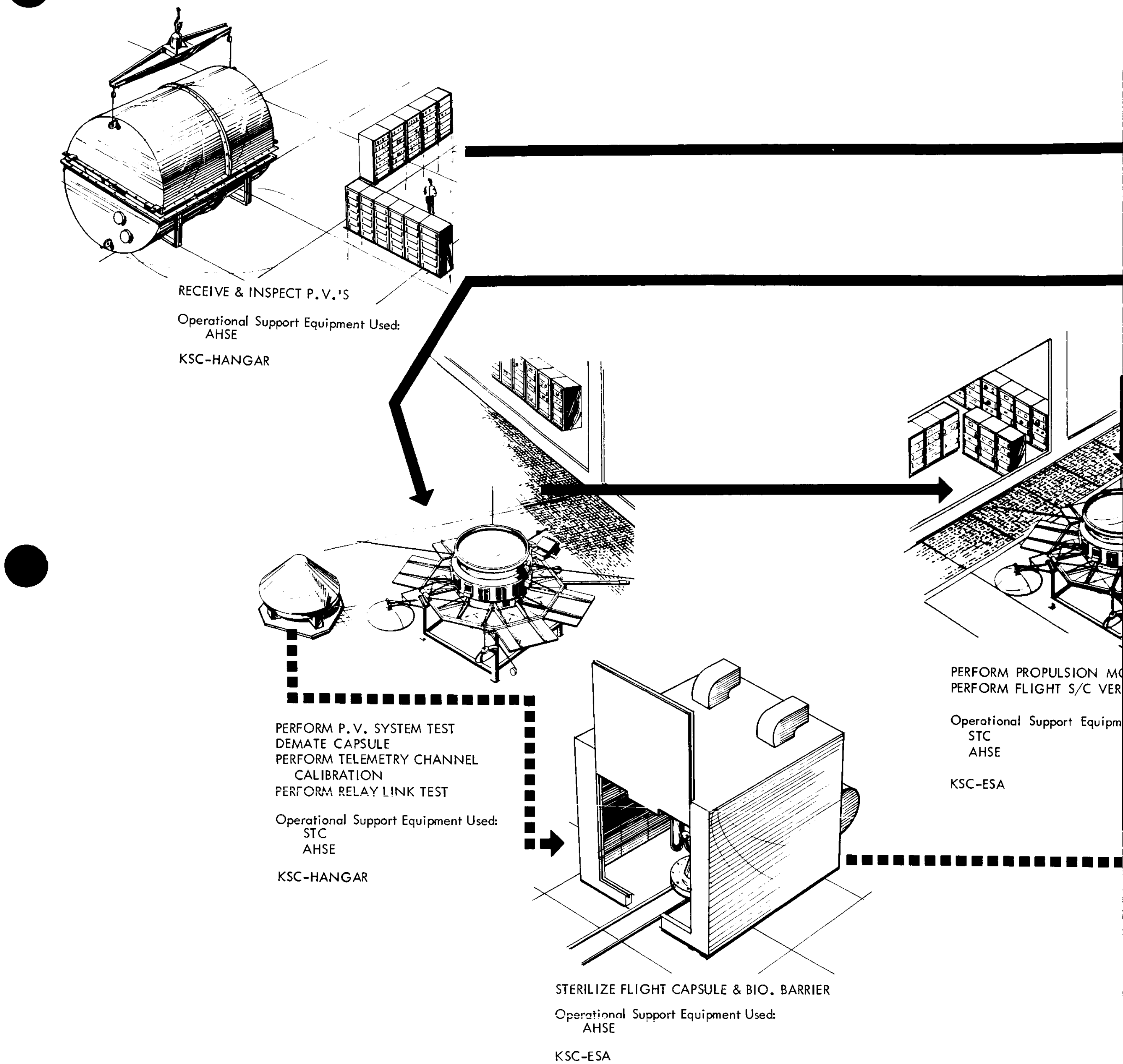
PERFORM FINAL P.V. INSPECTION
AND SYSTEM TEST
CONDUCT MISSION ACCEPTANCE REVIEW

Operational Support Equipment Used:
AHSE
STC
LCE

KENT ASSEMBLY & TEST FACILITY



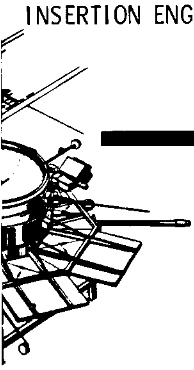
TRANSPORT
TO
KENNEDY
SPACE CENTER



RECEIVE & INSPECT P.V.'S
Operational Support Equipment Used:
AHSE
KSC-HANGAR
PERFORM P.V. SYSTEM TEST
DEMATE CAPSULE
PERFORM TELEMETRY CHANNEL CALIBRATION
PERFORM RELAY LINK TEST
Operational Support Equipment Used:
STC
AHSE
KSC-HANGAR
STERILIZE FLIGHT CAPSULE & BIO. BARRIER
Operational Support Equipment Used:
AHSE
KSC-ESA
PERFORM PROPULSION M
PERFORM FLIGHT S/C VER
Operational Support Equipm
STC
AHSE
KSC-ESA

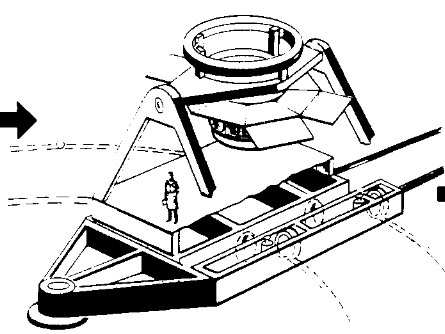
SIMULATED LAUNCH PROCESSI

PROVIDE LIVE
STERILE ORBIT
INSERTION ENGINE



)DULE LEAK TEST
IFICATION TEST

ent Used:

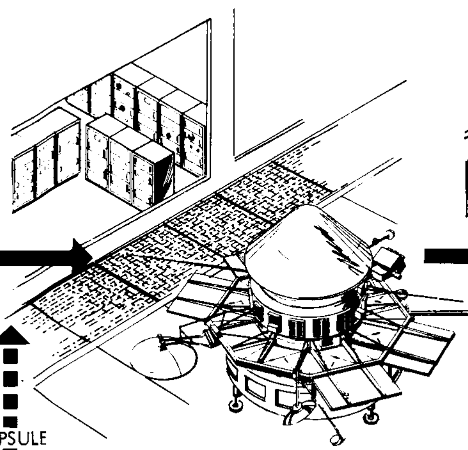


DETERMINE S/C WEIGHT & C.G.
MAGNETIC MAPPING

Operational Support Equipment Used:
AHSE
STC

KSC-ESA

CAPSULE



PERFORM P.V. VERIFICATION TEST
DETERMINE P.V. WEIGHT & C.G.
FINAL PROPULSION SYSTEMS ALIGNMENT

Operational Support Equipment Used:
AHSE
STC

KSC-ESA

FUEL S
DETER

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KSC-E

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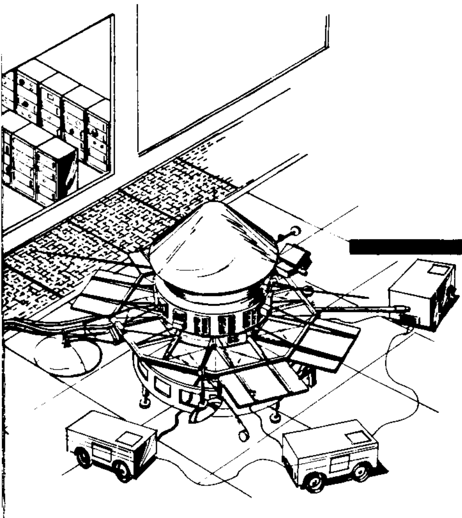
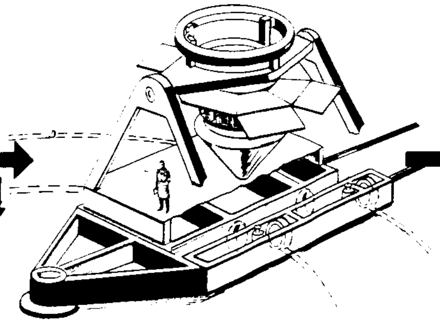
NG IDENTICAL TO BELOW



/C
MINE FUELED P.V. WEIGHT & C.G.

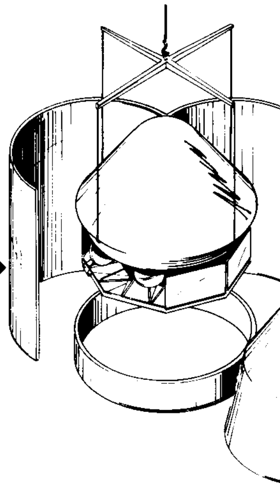
ional Support Equipment Used:
SE
C
SA



PERFORM P.V. MAGNETIC MAPPING
DETERMINE PAYLOAD MODULE WEIGHT & C.G.

Operational Support Equipment Used:
AHSE
STC

KSC-ESA



ASSEMBLE PAYLOAD
DETERMINE PAYLOAD MOD

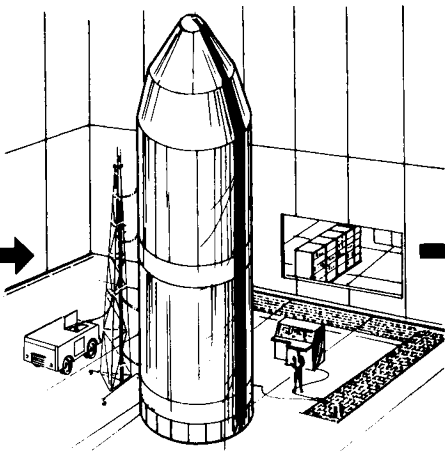
Operational Support Equipme
AHSE

KSC-ESA

3

ULE WT & C.G.

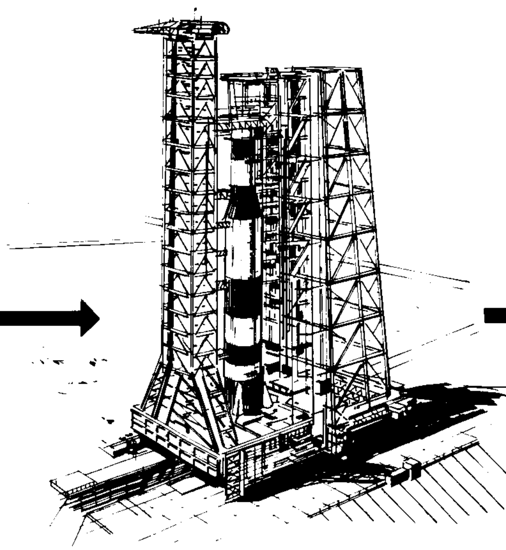
t Used:



PERFORM COOLING TEST
PERFORM SYSTEM INTERFACE TEST
PERFORM DUMMY RUN COUNTDOWN
PERFORM SIMULATED LAUNCH

Operational Support Equipment Used:
AHSE
LCE
MDE

KSC-ESA



PERFORM S/C-CAPSULE, MOS & TDS SYSTEM TESTS
PERFORM JOINT FLIGHT ACCEPTANCE COMPOSITE TEST
PERFORM SIMULATED LAUNCH

Operational Support Equipment Used:
AHSE
LCE

KSC-COMPLEX 39

COUNTD

Operation
LCE
MDE

KSC-COM

4

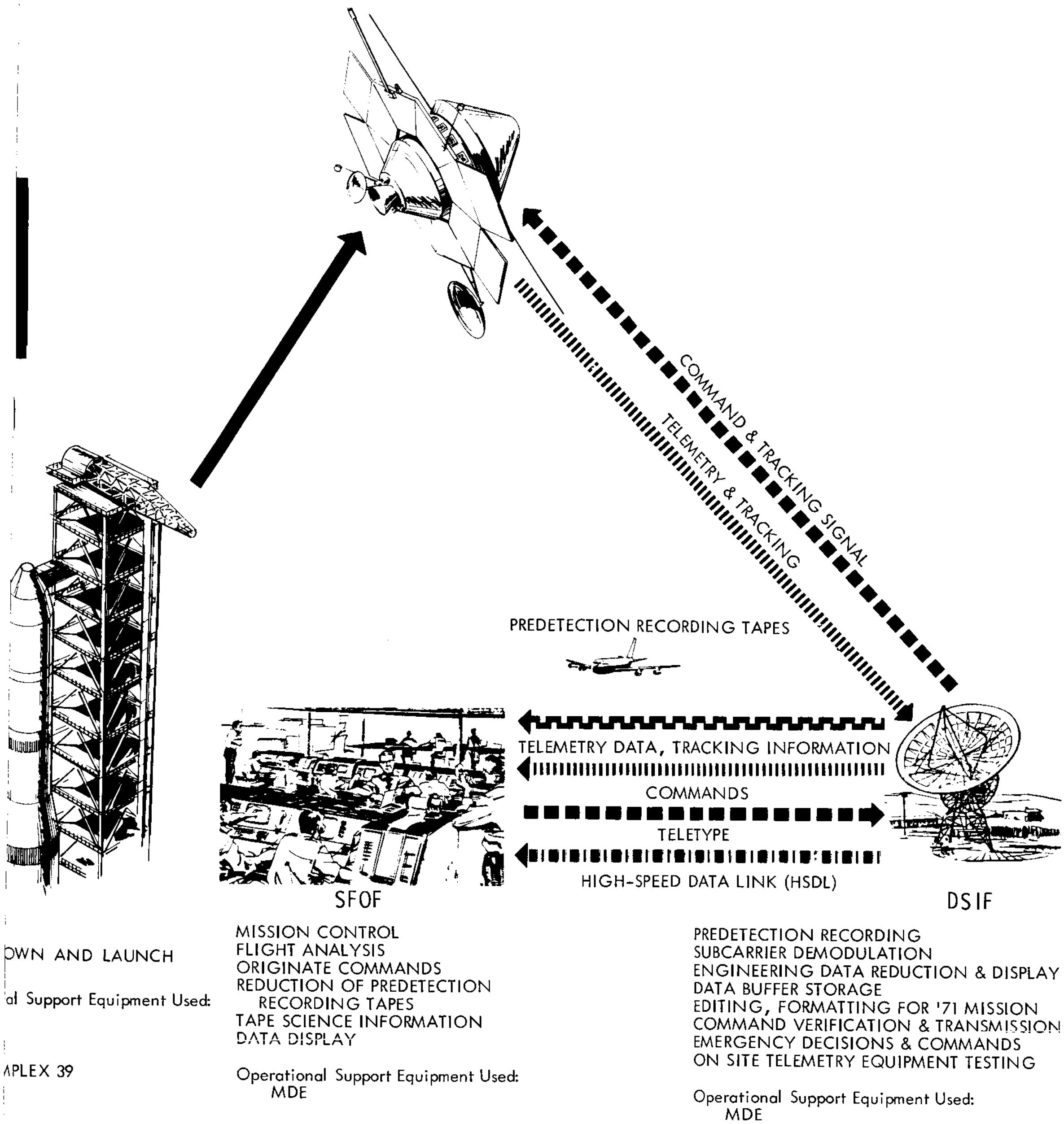


Figure 2.4-1: COMPOSITE SPACECRAFT - OSE OVERVIEW

Cape Processing--Operations are based on the concept of post-shipment tests, followed by a Planetary Vehicle processing dry run that duplicates the Planetary Vehicle final processing except for dummy ordnance and inert propellant. To provide further assurance of success, launch pad test operations are duplicated in the ESA to the maximum extent practicable before the encapsulated Planetary Vehicles with inert engines are taken to the launch pad and installed atop the launch vehicle. The live Flight Spacecraft orbit-insertion engine is installed in the final run. The STC equipment and personnel used to checkout the spacecraft at the factory are used for these operations.

The launch vehicle payload is transported to the launch pad and installed on the launch vehicle as three separate entities--an upper nose fairing and two Planetary Vehicle modules--to ease transportation and hoisting problems imposed by the total payload stack approach. The third Planetary Vehicle module remains at the ESA in a flight-ready condition for substitution in case of Planetary Vehicle failure.

Concept Refinement--Early in Phase IB, consultation with JPL is required to refine the concepts to: (1) provide increased emphasis on, and visibility to, activities that JPL experience has shown to be most critical on the Mariner and Ranger programs; (2) review sequencing to reflect JPL flight hardware scheduling; and (3) establish firm JPL/Boeing agreement on the program to provide qualified flight hardware and OSE at Cape Kennedy for the launch and acquisition of scientific data from Mars.

CONTENTS

3.0 SYSTEM-LEVEL OPERATIONAL SUPPORT EQUIPMENT

System-level OSE includes the System Test Complex (STC), Launch Complex Equipment (LCE), Assembly, Handling and Shipping Equipment (AHSE), special test facilities (STF), Mission-Dependent Equipment (MDE), and associated software.

3.1 SYSTEM TEST COMPLEX

Major elements of the System Test Complex include:

- Appropriate portions of spacecraft bus and propulsion subsystem OSE, e.g., only three of the nine items making up the power subsystem OSE are required at the system test level. Section 4.0 of this document discusses this point as applicable to each of the subsystem support equipments.
- Government furnished science subsystem OSE.
- Test set adapters.
- A central data and control system.
- A voice communication system.
- Simulators (GFE) for the launch vehicle, science subsystem and capsule.
- Cabling.

3.1.1 Summary

The Flight Spacecraft STC performs the system-level functional testing of the Spacecraft Bus and the complete Flight Spacecraft, and, in conjunction with the government furnished capsule STC, performs the Planetary Vehicle testing. In addition, the spacecraft STC is used to support electromagnetic interference (EMI) tests, environmental tests, and magnetic mapping.

Figure 3.1-1 shows a model of the STC in a typical test arrangement. It consists of approximately 75 racks and 1 console, 8 of the racks being the estimated Science Subsystem OSE and 14 racks for the capsule test equipment. See Table 3.1-1 for SS OSE equipment quantity and use.

To ensure continuity of subsystem test data, the appropriate elements of test equipment for each of the Spacecraft Bus subsystems and the Science and Propulsion subsystems are integrated into the STC and used during system testing. This allows optimum correlation to be made between the various test data obtained in the several stages of testing. For STC use, the test sets are connected not only to the individual subsystems but also to a group of test-set adapters. The adapters convert the test sets into units that interface between the Central Data and Control System and the spacecraft. To properly sequence and interleave the various tests and to display and record test results, the automated Central Data and Control System is used to control the test process. Surveillance and supervision over this data system is exercised by the test conductor and his staff who are provided with the capability to interrupt or modify the sequence of computer routines that program the accessing, processing, and distribution of test data.

The STC is supported by its own power, environmental control, and voice communications facilities.

3.1.2 Applicable Documents

Basic reference documents applicable to all OSE design are listed in Section 2.1. A list of documents of specific concern to STC design follows.

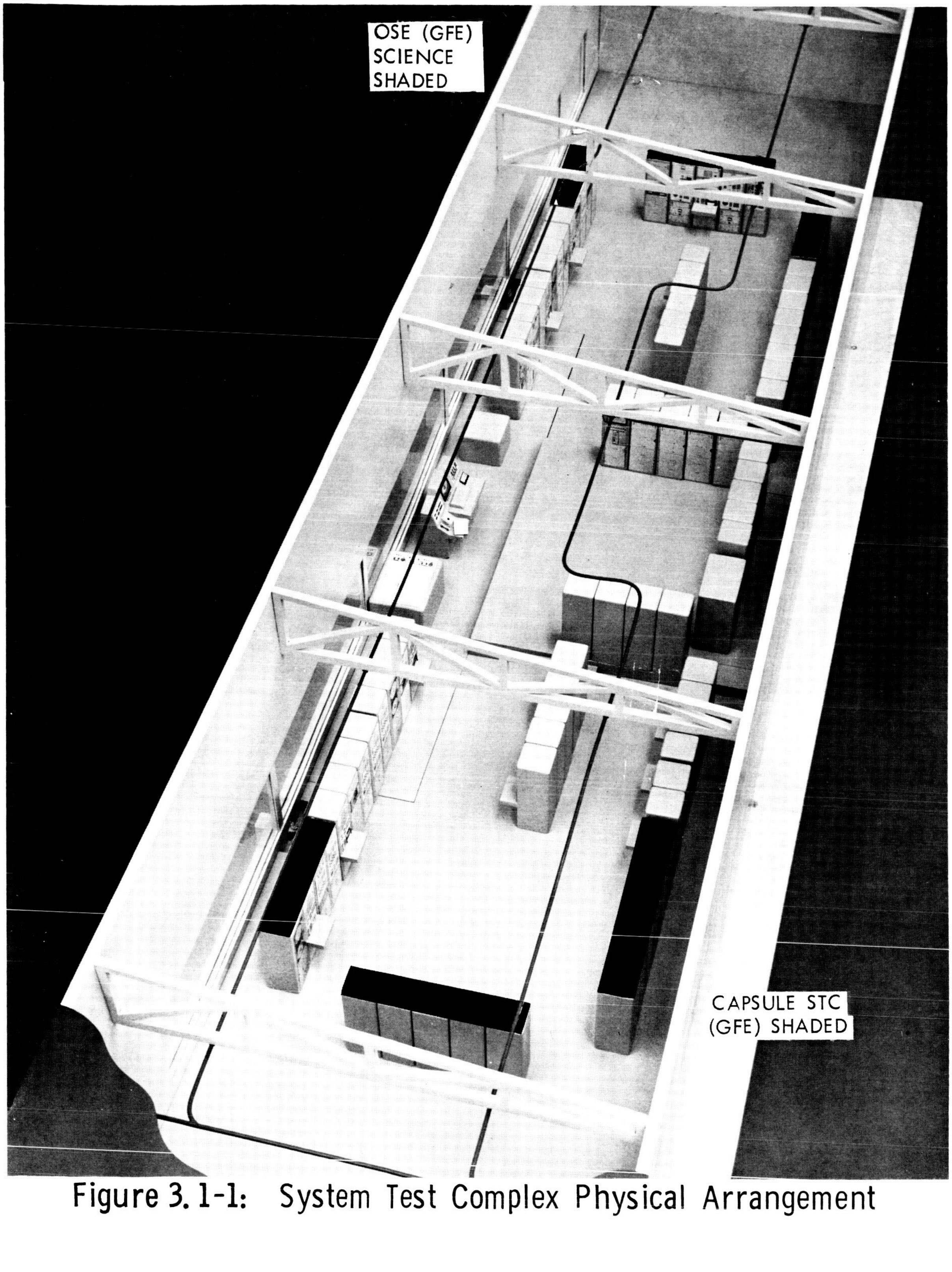


Figure 3.1-1: System Test Complex Physical Arrangement

Table: 3.1-1 SS OSE EQUIPMENT

SS OSE AND TEST COMPLEX ELEMENTS	TOTAL RACKS* OF SS OSE	EQUIPMENT USED IN STC				
		DATA REDUCTION & DISPLAY AREA			REMOTE AREAS	
		SS OSE RACKS	SSTS ADAPTER RACKS	CONSOLES	SS OSE RACKS	SSTS ADAPTER RACKS
Power SS OSE	9	3	1		2	
Guidance & Control SS OSE	4	2	1		1	
Data Storage SS OSE	4	2	1			
Radio SS OSE	7	6	1			
Telemetry SS OSE	5	5	1			
Command SS OSE	3	2	1			
Computer & Sequencer SS OSE	2	2	1			
Structures & Mechanisms SS OSE	2		1		2	
Pyrotechnics SS OSE	2	1	1		1	
Temperature Control SS OSE	1	1	1			
Propulsion SS OSE	5				5	1
Science SS OSE**	14	8	1		2	
Central Data & Control System	18	18		1		
Capsule STC**	14	14				
TOTALS	90	64	11	1	13	1
SUMMARY	90 Racks* of SS OSE	75 Racks* & 1 Console Used in STC Data Reductions & Display Area			14 Racks* Used in STC Remote Areas	

*Actual or equivalent racks
**Figures are estimates of GFE items

1) JPL OSE/MC Specification Document (Phase IA, Task B Statement of Work Support Data).
2) OSE/MC-4-130-A, "Functional Specification Mariner C, OSE Computer Data System (JPL)."
3) Drawing 890 1550, OSE Cabling Block Diagram (JPL).
4) "Test and Operations Plan, Mariner C," October 30, 1963.

3.1.3 Design Constraints and Requirements

The fundamental constraints and requirements governing the derivation of the System Test Complex OSE design are those contained in the documentation that accompanied the Phase IA, Task B statement of work (see Section 2.0). These are listed in Section 3.1.4 because of their major influence on the rationale for the STC recommended concept described and compared in that section. The means of ensuring STC design compliance with the constraints and requirements is described below.

3.1.3.1 Compliance

A diagram depicting the process followed in ensuring compliance is presented in Figure 3.1-2. In essence, the process was one of:

1) Identifying every statement of potential impact on the STC;
2) Evolving functional flow charts depicting the process of assembly and test for the Spacecraft System flight hardware (see Section 2.4);

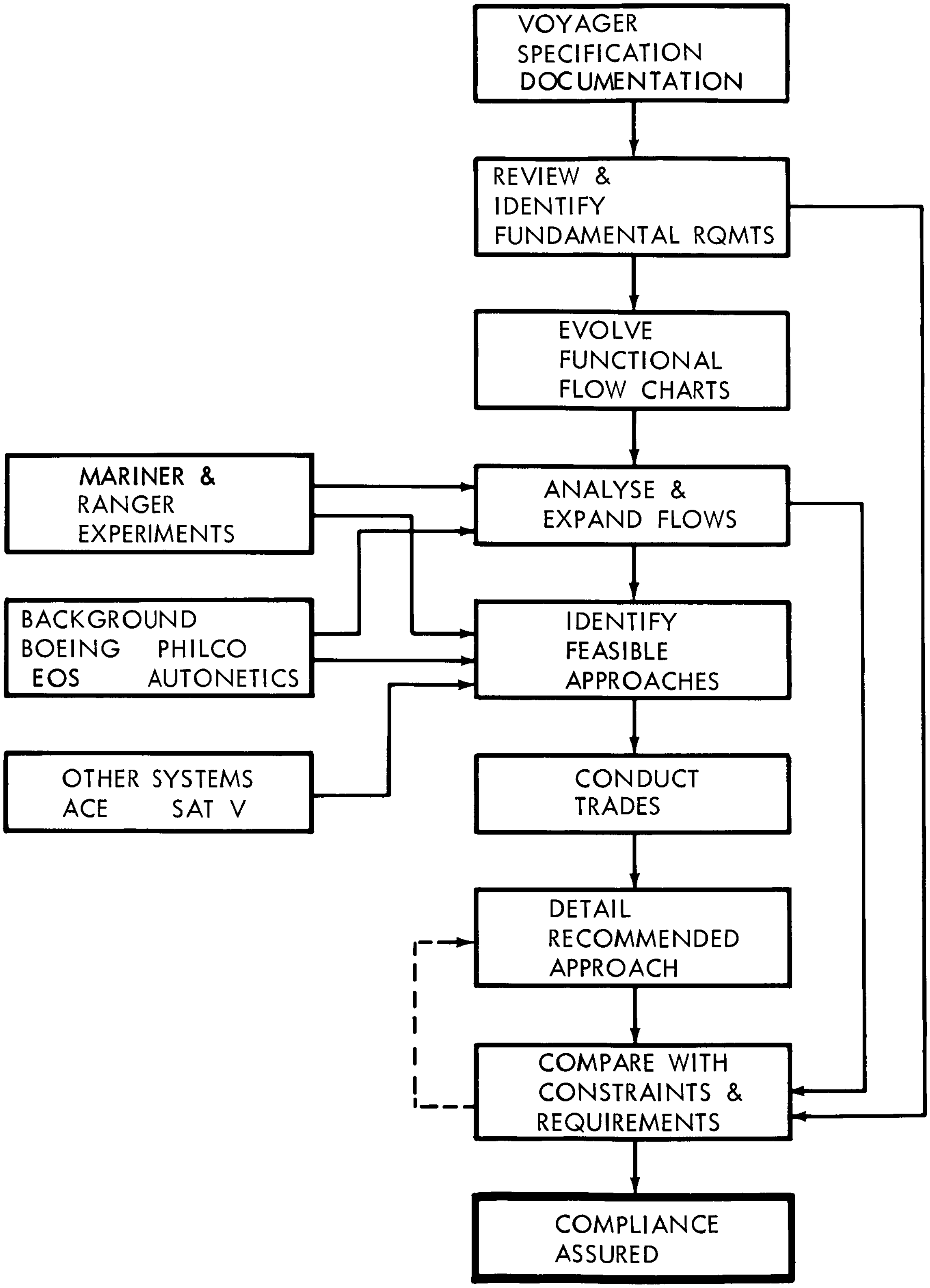


Figure 3.1-2: Requirements Compliance Process

3) Determining what the STC must do and where it will be used. (See Section 3.1.3.2);
4) Identifying how the similar tasks were accomplished by Ranger and Mariner;
5) Identifying the feasible approach to the tasks and conducting trades to select the most desirable (see Section 3.1.4);
6) Developing a preliminary design of the selected approach, comparing this design with the requirements, and reviewing where necessary;
7) Conducting design reviews and finalizing the design.

The end result is an STC design incorporating the fundamental and derived design constraints and requirements in keeping with the logical evolution of the Voyager Program from the Ranger and Mariner experience.

3.1.3.2 Derived System Requirements

STC requirements, in addition to those given in the preliminary specifications, have been derived through a system analysis and expansion of functional flow charts. The test location usage of STC equipment for spacecraft acceptance testing is shown in Figures 3.1-3 and 3.1-4. These flow diagrams are derived from the processing flow shown in Section 2.4. As shown, the STC is required to:

1) Perform Spacecraft Bus, Flight Spacecraft, and Planetary Vehicle testing at the Kent, Washington, general-purpose test area;
2) Support performance of EMI, vibration, magnetic-mapping, thermal-vacuum, free-mode, and payload testing at Kent;

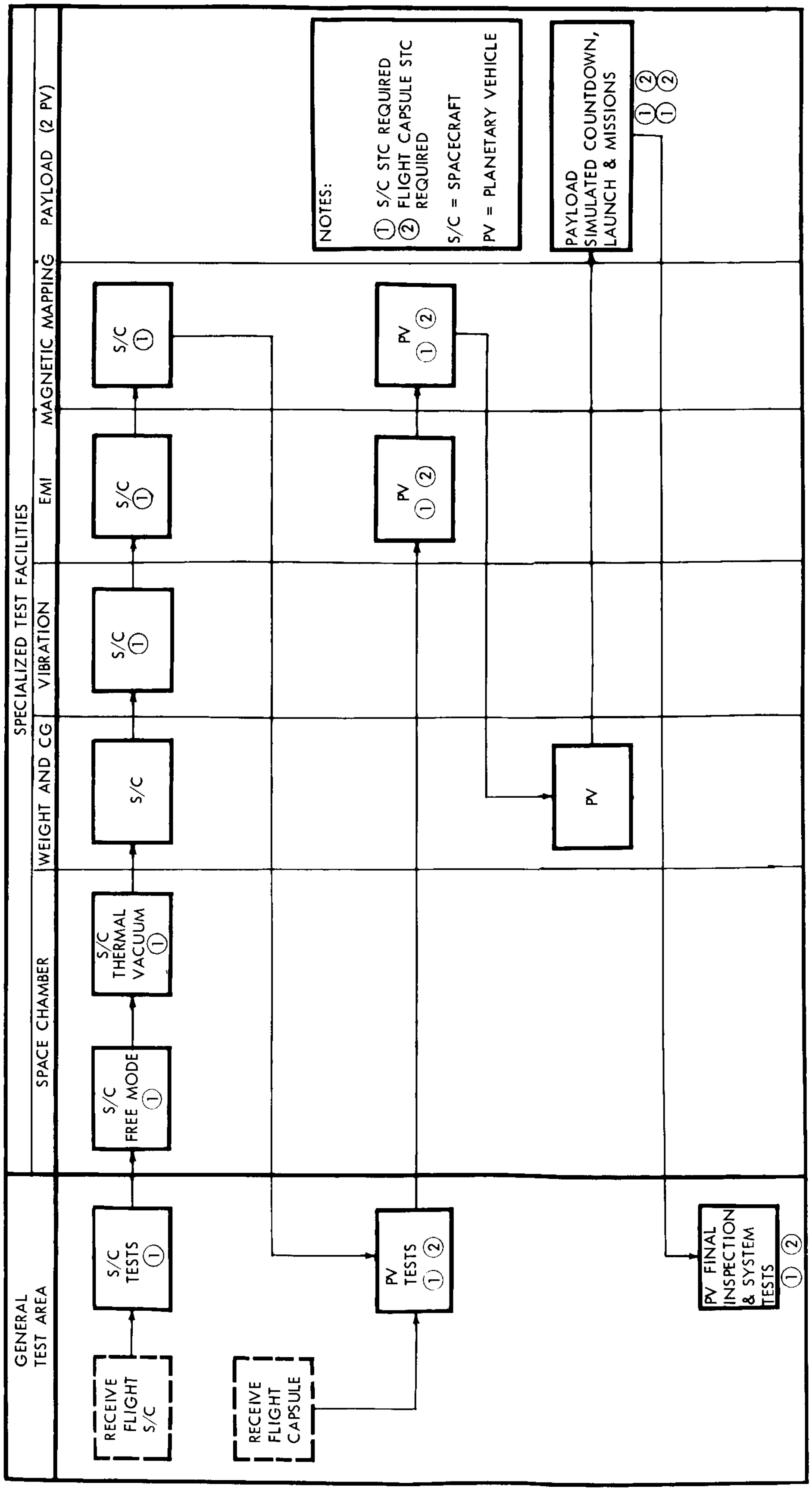


Figure 3.1-3: STC Usage — Kent Space Center Processing

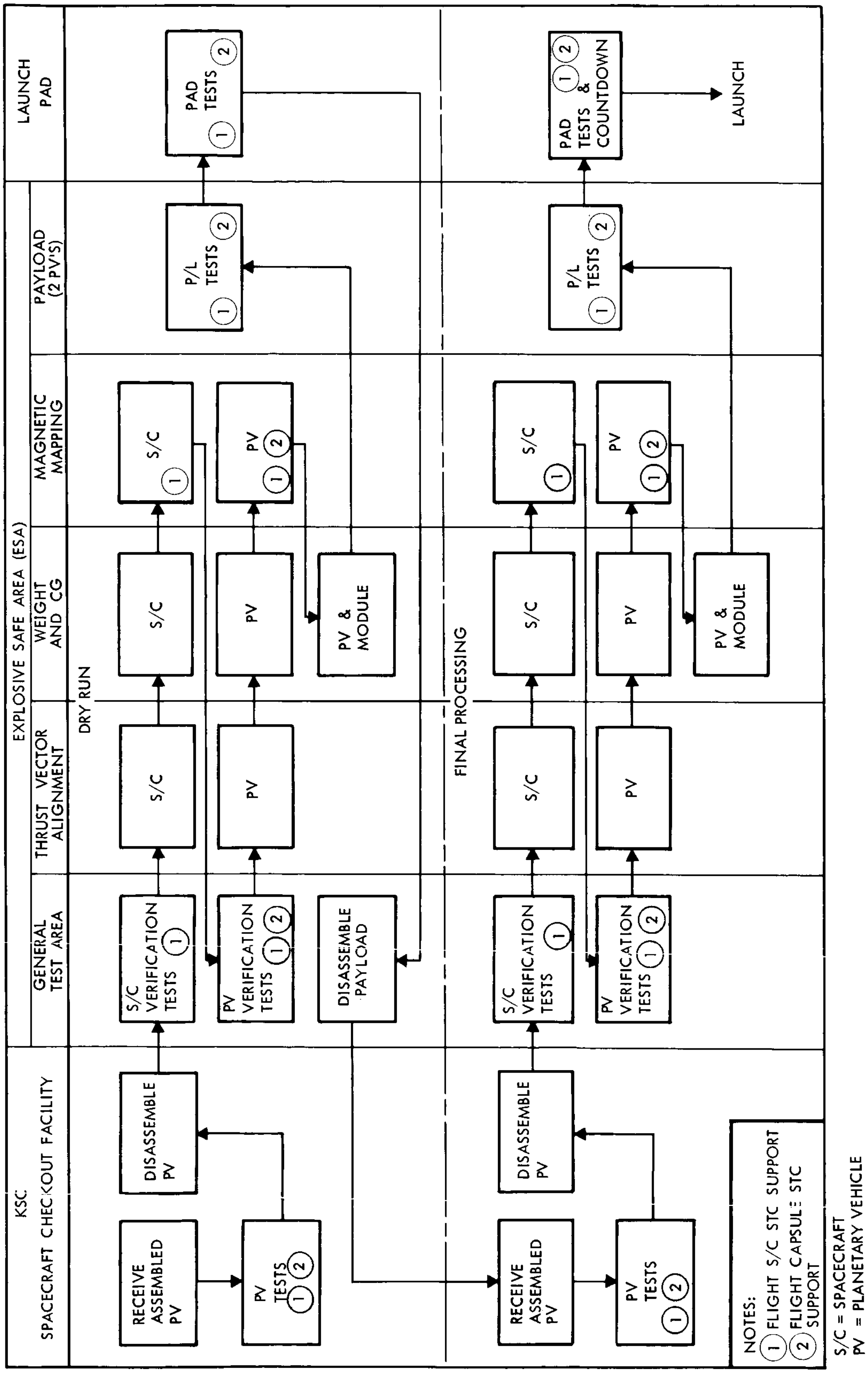


Figure 3.1-4: STC Usage — Kennedy Space Center Processing

3) Perform Flight Spacecraft and Planetary Vehicle functional tests at the spacecraft checkout facility at Kennedy Space Center (KSC);
4) Perform Planetary Vehicle functional tests at the explosive safe area (ESA);
5) Support performance of magnetic-mapping tests at the ESA (KSC);
6) Support LCE in the performance of payload tests at the ESA;
7) Support LCE in the performance of launch pad checkout and countdown operations.

3.1.4 Trade Studies

The Phase IA, Task A Voyager study described an STC compatible with a single spacecraft using a Saturn-IB/Centaur launch vehicle. The Phase IA, Task B study proposed the simultaneous launch of two Planetary Vehicles employing a Saturn V launch vehicle. In view of these changes, a re-evaluation of the STC was required. Several existing test systems were considered for modification and use in the Voyager STC and were traded-off against an updated and revised Task A preferred concept.

3.1.4.1 Functional Requirements

All candidate systems were evaluated against the JPL requirements and constraints. In addition, STC derived requirements were used in the comparison. These requirements are as follows:

1) The SS OSE, which is used for subsystem testing, will form the basic intertie between the Spacecraft Subsystems and the STC.
2) The alarm and safety circuits associated with the individual SS OSE will be directly connected to an event logic network so that priority-interrupt technique can be employed.

3) The STC will be employed during system testing and in support of the launch operations. In the launch mode, capability to independently test two Flight Spacecraft simultaneously is required.

4) The STC will be capable of driving parallel subsystem displays at the SS OSE during system testing.

5) The commercial computers selected should be compatible with the JPL computer facilities and the DSIF network facilities.

3.1.4.2 System Concepts

The above requirements and constraints reduce the number of feasible concepts to three. The three represent the state of the art in aerospace test systems and provide sufficient information for a satisfactory comparative evaluation. They are a modified acceptance checkout equipment - Apollo (ACE) system, and updated Mariner C system, and the Phase IA Task B system. The ACE system being developed for the checkout of Apollo manned systems could be made available for the Voyager 1971 Program. The Mariner C concept--an earlier, more conservative JPL design--has the data-acquisition capability required and has limited test capability, but must have command capability added to satisfy the system requirements.

The Boeing Task B Voyager concept reflects a modest evolution in automaticity from the JPL Mariner concept and includes automated command capability. It still retains conservatism in its functional design, but provides for substantial flexibility and growth potential.

Other test systems, similar in concept to those discussed here, are tabulated below, with the company or agency that developed each and the vehicle or system with which each is to be used.

ESE	Boeing	Saturn S-IC
VATE	Hughes	General purpose
ACRE	Lockheed	Polaris
TEMS	Martin	Mace
OMTS	JPL	Sergeant
---	STL	OGO
D-PAT	Hughes	General purpose
MAIDS	Frankford Arsenal	Field maintenance
SE-1000	Packard Bell	General purpose

ACE System--Figure 3.1-5 is a block diagram illustrating how a modified ACE system could be used as the STC for Voyager. The operation of the modified ACE system begins with the start modules in the control and display area associated with particular subsystem test sets. The start modules permit the operator to issue discrete commands to the test set, to initiate a 4-data-bit word that addresses a particular subroutine in the command computer, and to read a punched paper tape routine into the system. The communications unit executor (CUE) scans all start modules for an "Execute" signal; when one is found, it accepts the information and routes it to the computer along with the address from which it came. The command computer receives the address data words from the CUE and performs the required processing. If a command is to be sent, it is processed by the data transmission and verification converters (DTVC's).

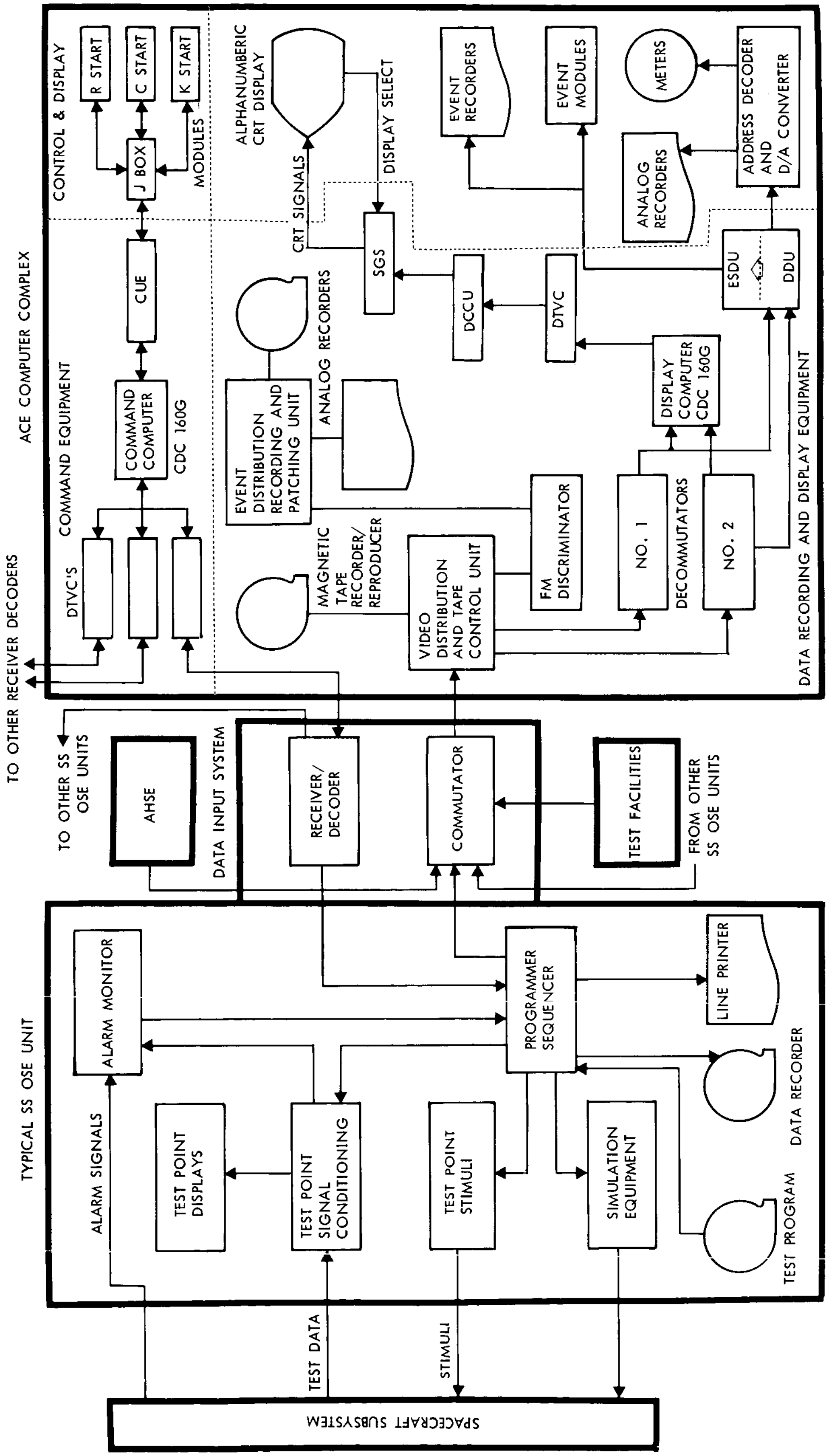


Figure 3.1-5: Modified ACE

The DTVC is a two-way communicator and a parallel-to-serial and serial-to-parallel converter. All computer input and output communications are in parallel format, but all transmissions to and from the receiver-decoder at the spacecraft test area are in serial format.

The receiver-detector, which is not part of the ACE spacecraft station, receives the first 24-bit word from the DTVC and assembles the first 12 bits in half of a 24-position register. The receiver-decoder performs the first level of address decoding, and leaves 9 to 10 bits for control of the subsystem test set or sets.

Outputs of the subsystem test sets are frequency modulated (FM) or serially pulse-code modulated (PCM) formatted and sent into a command commutator. The output is sent via hard lines back to the ACE spacecraft station.

The incoming data lines are initially terminated at the distribution and tape transport control unit, which provides patching and switching to route the data to the desired decommutator, discriminator, and magnetic-tape recorder and reproducer. The data are routed as follows:

1) All data are routed to wideband magnetic-tape recorders.
2) The FM data can be routed to discriminators and then to an analog recorder.
3) The PCM data are routed to one of two decommutator units.

The decommutator unit converts the incoming 8-bit serial data words into parallel format and addresses them for distribution. The decommutator output goes to the display computer and the decommutator distribution unit.

The display computer performs the following primary operations:

1) Assembles the data into proper format and length;
2) Compares data to predetermined limits and flags out-of-limit conditions;
3) Computes the average of a block of words and retains this average for output display;
4) Converts data into engineering units using curve-fitting techniques;
5) Converts data and places it in proper format for alphanumeric cathode-ray tube (CRT) display.

The symbol generating system (SGS) unit receives the coded words from the digital communications conversion unit (DCCU). The 12-bit words, each representing two display characters, are stored in memory locations according to received instructions and addresses. Memory allocations in the SGS are sufficient to provide for display of 20 pages of data.

The alphanumeric display system displays analog measurements and certain event occurrences. It provides simultaneous viewing of 24 lines of data on a CRT. Switching at the display permits the call-up of 40 separate 12-line data tabulations with suitable function and page identification.

Event data words are selected from intermixed analog and event data words received from the decommutator distribution circuits in the event storage and distribution unit of the digital data unit/event storage and distribution unit (DDU/ESDU). The ESDU will select and store 150 event words, each of which consists of 8 discrete event functions (a total of 1200 events). The outputs of the storage registers are routed to a patch facility in the DDU/ESDU. The patching at this point determines the distribution of each event function to the various consoles and permits routing any specific event function to multiple locations.

Mariner C STC System--Previous JPL studies have shown that the STC should consist of an integrated set of test equipment specifically designed to support the system testing of the spacecraft. Each subsystem test set should be designed to support the testing of an individual subsystem during subsystem and system test operations. A computer data system capable of processing both telemetry and hardwire data is desirable. These design criteria were included in the OSE for the Mariner C program. The resulting automatic data system for system test application contains a medium-sized, high-speed computer to process both telemetry and hardwire data from the Spacecraft Systems. The hardwire input data are of several types: (1) analog, (2) events, (3) ac-voltage or pulse inputs for counting, and (4) serial or parallel data-register inputs. The system block diagram (Figure 3.1-6) outlines the major system components.

The major innovations of these systems in JPL Spacecraft System testing are as follows:

1) All telemetry and hardwire data samples are permanently recorded on digital tapes for off-line data processing as required.
2) All classes of data can be tabulated in engineering units in real time if required.

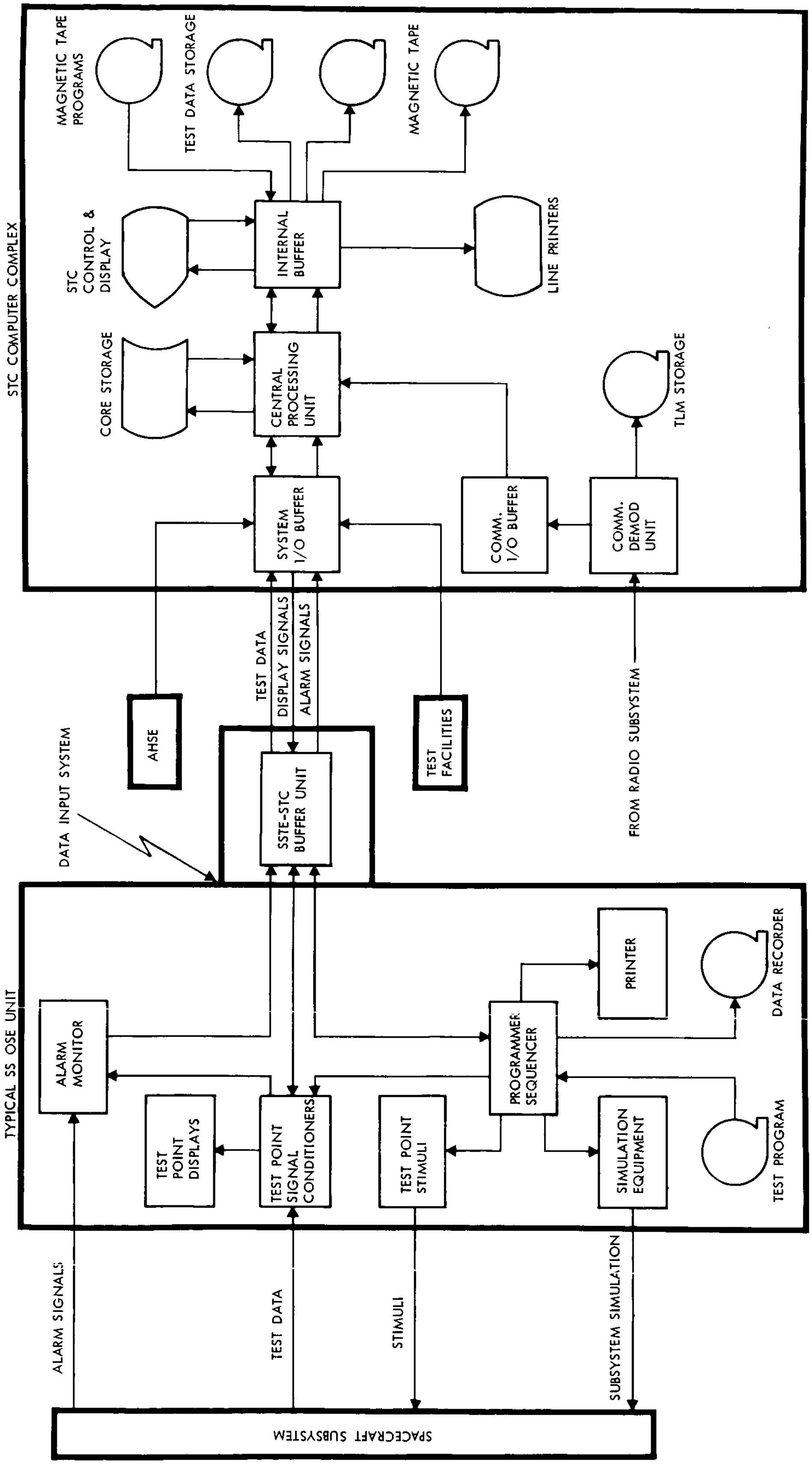


Figure 3.1-6: Mariner C Configuration

3) Automatic alarm-limit checking will warn of failures.

4) Abbreviated printouts, wherein only data that have changed since the last printout will be provided in real time.

5) The limited hardwire data that are presently being acquired on an analog oscillograph (central recorder) will be recorded, limit-tested, printed, or processed to provide information to supplement that of the telemetry channel.

Each data system includes two data input subsystems, each located in the STC near the spacecraft and SS OSE, and a centrally located computer associated with the facility. Cable drivers are provided so that two data input subsystems, together with the associated ground support equipment (GSE) area data printers, can be operated at a remote location. There are five data printers in each test complex--a line printer for systems analysis, and four character printers (typewriter-like printers) serving the following technical areas: (1) telemetry and communications; (2) science; (3) power, central computer and sequencer (CC&S), and attitude control; and (4) temperature control, pyrotechnics, and propulsion.

Each data system performs the following tasks:

1) Scans the output of one or two telemetry data systems (communications buffer);

2) Scans the outputs of one or two data-input subsystems, each of which contains 100 analog input channels, 50 on-off-type event channels, and 6 counters (system input-output buffer);

3) Digitally records all data samples in raw form, including suitable time tags and channel identification (internal buffer--magnetic tapes);

4) Records raw telemetry mixed subcarriers or pulse codes on analog magnetic tape (telemetry magnetic tape);

5) Records selected hardwire data on analog magnetic tape or on the central recorder (oscillograph);

6) Processes data: (a) alarm-limit checks all data samples from selected telemetry, hardwire, or counter channels; (b) reduces data from all channels (telemetry or hardwire) to engineering units (not all samples in each channel are so reduced); (c) prints out all data channels on a line printer in each system data analysis area; (d) prints out selected (usually preprogrammed) data on four GSE-area character printers located in each test complex; and (e) prints event and alarm-limit messages on selected printers as dictated by the source of the alarm information;

7) Provides non-real-time 11- by 17-inch plots such as telemetry channel calibrations, cross plots, or time plots of recorded data;

8) Provides digital magnetic tapes for various on-line or off-line data analyses (IBM-compatible digital magnetic-tape records will permit the Space Flight Operations Facility (SFOF) IBM 7090 computer facilities to be used for trend analysis and statistical analysis of data samples. Plots can also be provided by the SC 4020 or other SFOF data display equipment. SFOF-compatible recordings will make the data available to flight-analysis teams through the SFOF system);

9) Provides a limited amount of command capability.

Phase IA, Task B STC--This STC is an operational unit as diagrammed in Figure 3.1-7 and described in Sections 3.1.5 through 3.1.10.

3.1.4.3 System Comparison

The following factors were chosen as the basis for trade studies:

1) Agreement with previous JPL STC concepts;
2) Compliance with Voyager STC functional requirements;
3) Compliance with the STC constraints;
4) Degree of modification of existing equipment required;
5) Additional constraints introduced;
6) Capability for rapid disassembly, transportation, and assembly.

The comparative evaluation of the three concepts with these parameters is tabulated in Table 3.1-2.

3.1.4.4 Conclusions

The results of the concept comparisons follow.

1) Integrating the subsystem OSE into the ACE is required for manual control capability and common equipment usage throughout subsystem and system testing.
2) Integration of the subsystem OSE into the ACE system results, in some instances, in redundant automaticity.
3) The modified ACE system has the capability to perform the desired tests.
4) The modified ACE does not meet the need for expeditious disassembly, transport, and assembly.

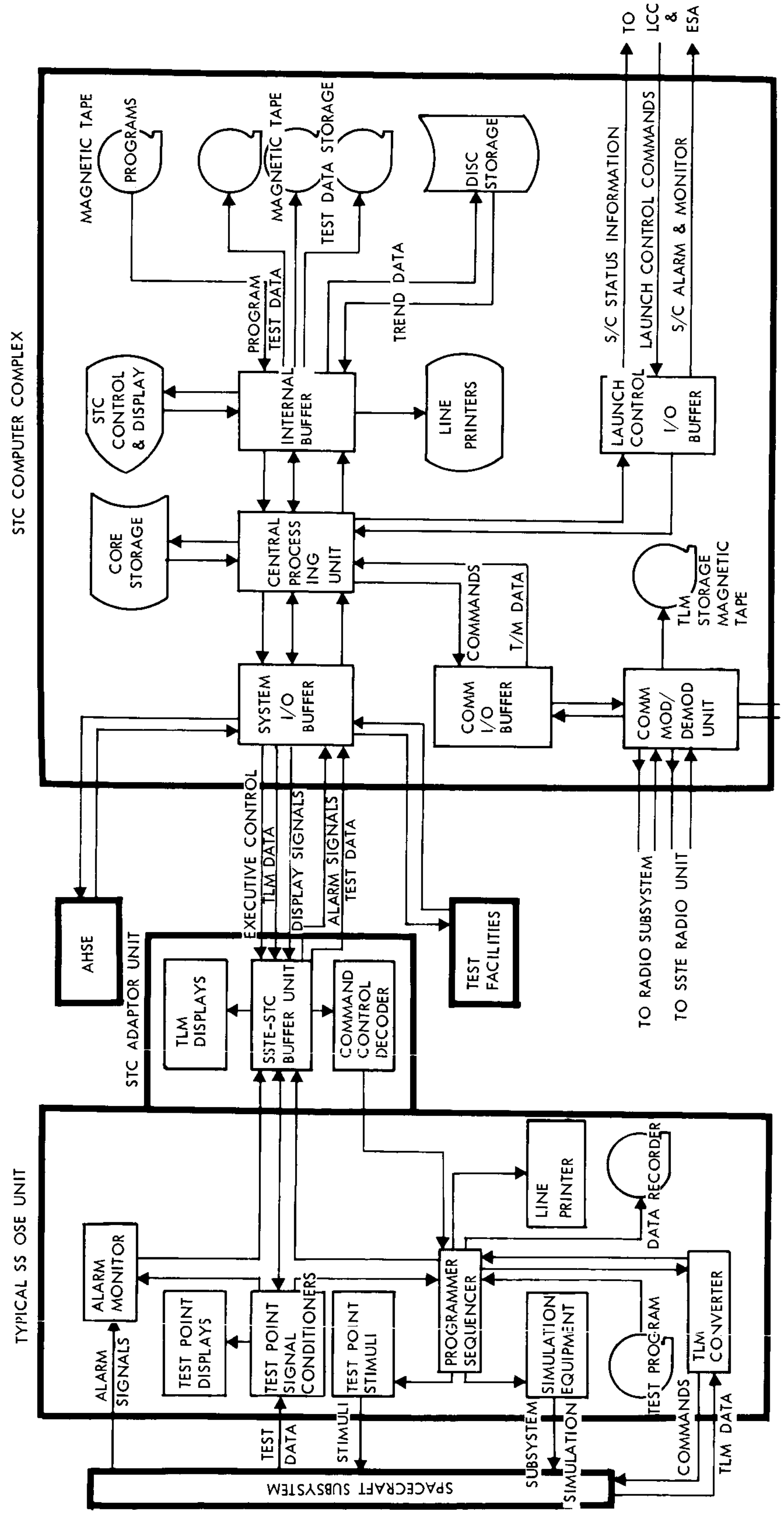


Figure 3.1-7: Phase IA Configuration

Table 3.1-2: SYSTEM COMPARISON TABULATION

TRADE FACTORS	MODIFIED ACE	MARINER C	PHASE IA
1) Agreement with previous JPL STC concepts.	The approach does not evolve from previous JPL practice.	Basic JPL concept.	Evolves from JPL practice.
2) Compliance with Voyager STC functional requirements.			
a) Complete testing of individual and collective subsystems.	Yes -- Possible through use of the programmable ACE subsystem control consoles.	No -- Not sufficiently developed to permit the desired level of complete testing.	Yes -- Implementation of the advanced operational support equipment and data processing provides the desired capabilities.
b) Implement repeatable, accurate, and expeditious test routine.	Yes -- The degree of automation of test routine data storage and recall provide this capability.	No -- The degree of time independency for sequencing and control is not adequate.	Yes -- The use of the independent monitor and control mechanisms provides this capability.
c) Provide for both performance and margin testing.	Yes -- Can be programmed as required.	No -- Insufficient central control in the Univac 1218 computer.	Yes -- Performance tests can be implemented from either the central data processor or subsystem OSE; margin tests are done via the subsystem OSE.

Table 3.1-2: SYSTEM COMPARISON TABULATION (continued)

TRADE FACTORS	MODIFIED ACE	MARINER C	PHASE IA
d) Provide for both central and manual control.	Yes -- Use of the Voyager subsystem OSE and computer center provides both.	No -- Equipment limits and time make unpractical.	Yes -- But with somewhat less central control as compared to the ACE system.
e) Provide for both central and local data processing capabilities.	Yes -- Programmable as required.	No -- Real-time multiple hard copy printouts restricted and data suppression techniques are used.	Yes -- Programmable as required.
f) Fault isolation to replacement level.	Yes -- Manual and automatic modes are available.	No -- Insufficient control capability to isolate faults to module level.	Yes -- Manual and automatic modes are available.
g) Flexibility to accommodate configuration changes.	Yes	Yes	Yes
3) Compliance with the STC constraints			
a) Only commercial computer equipment will be employed.	Yes -- The CDC-160G is commercially available and proven.	Yes -- Univac 1218 is commercially available and proven.	Yes -- The SDS 900 series is commercially available and proven.

Table 3.1-2: SYSTEM COMPARISON TABULATION (Continued)

TRADE FACTORS	MODIFIED ACE	MARINER C	PHASE IA
b) Only general-purpose computers will be used for central control and data processing.	Yes	Yes	Yes
c) Provide compatibility between OSE computer modules.	Yes	Yes	Yes
d) Use a standard approach to computer selection.	Yes	Yes	Yes -- The SDS 900 series has been selected for technical reasons.
e) Provide programming compatibility through all test sequences.	Yes -- Items c and d.	Yes -- Items c and d.	Yes -- Items c and d.
f) No new test language will be developed.	Yes -- Existing test language (ATOLL) will be used.	No -- Programmable only in machine language.	Yes -- Existing Fortran II and symbolic language are available.
4) Degree of modification of existing equipment required.	All modularized subsystem control consoles.	Implementation of an improved test control system.	None.

Table 3.1-2: SYSTEM COMPARISON TABULATION (continued)

TRADE FACTORS	MODIFIED ACE	MARINER C	PHASE IA
	Apollo software test routines.	Development of a command up-link system.	
	The 12-bit control logic output of the receiver-decoder.	Expanded control and data processing capabilities.	
	The commutator data input would change to accept hardline data.	Separate MDE checkout facility required.	
	The Voyager telemetry data frame and word format would require a change in the programming of the decommutator.	NOTE: With these modifications, the Mariner C system and the Voyager Phase IA concepts are alike.	
	Three input/output channels must be added to the command computer.		
	The base plates interfacing between the receiver-decoder output and the spacecraft would be replaced.		

Table 3.1-2: SYSTEM COMPARISON TABULATION (continued)

TRADE FACTORS	MODIFIED ACE	MARINER C	PHASE IA
	Digital data transmission lines must be installed between MILA and the JPL test facilities. Separate MDE checkout facility required.		
5) Additional constraints introduced	Further analysis to determine: (a) if the 12-bit command word imposes restrictions on initiating tests or processing data, and (b) the impact on the data processing system of eliminating the carry-on equipment.	None if the above modifications are implemented.	None
6) Capability for rapid disassembly, transportation, and assembly	Complexity of system appears to make transportability poor. Design was based on a fixed facility concept.	Yes -- Used effectively on Mariner C.	Yes -- This is an updated Mariner system.

5) The Mariner C concept does not meet all of the Voyager specifications.

6) The modified Phase IA, Task A STC design concept is a natural evolution of the basic Mariner concept above and meets all the Voyager specifications.

7) The SS OSE assemblies provide a common programmable test capability that could be used with all of the SS OSE sets. Specifically, these include:

   a) Data printer;
   b) Test program tape reader;
   c) Data processor logic
      (1) Buffer and command decoder,
      (2) Input register control and word transfer,
      (3) Digital comparator,
      (4) Test point selector;
   d) Control panel.

8) A number of general-purpose computers available commercially could be satisfactorily used in the STC. Specifically, these include:

   a) PB-450;
   b) SDS-920/930/9300;
   c) CDC-160G;
   d) GE-225;
   e) Univac 1218/1219;
   f) PDP-4.

9) An SDS-900 series computer has the unique advantages of:

   a) Functional capabilities that could provide the checkout facility for all DSIF MDE software programs.

b) Potential to enhance the flight simulation test by providing a real DSIF data processing capability.

3.1.4.5 Recommendations

Based on the comparison, the selected concept is a modified version of the Boeing Phase IA, Task A system. The difference between the Mariner C concept and this Boeing Task B design (Task A concept modified) is the degree of automatic system testing and control employed. The Task B STC includes the basic data acquisition and processing capability of the Mariner system plus the required control capability via either the SS OSE unit or the radio command system. This system is the natural evolution of the Mariner test system, and is based on conservative upgrading of the Mariner STC concept.

3.1.5 Selected STC Configuration

The STC, which consists of five major hardware elements, provides the system-level test capability required for performing the tests identified in Section 2.4. The five major elements are:

1) Subsystem test sets (SSTS);
2) STC/subsystem test set adapters (STC/SSTS adapters);
3) Central data and control system (CDCS);
4) Voice communications system;
5) Simulators.

The associated test and checkout computer programs are discussed in Section 3.6.4.2.

3.1.5.1 Functional Overview

The detailed interconnection and functional relationship of the above elements are dependent on the particular test being conducted. Therefore, a representative sample test configuration is used as a basis for this functional overview discussion.

Sample Test Configuration--

1) Science Subsystem mated to Spacecraft Bus;
2) Subsystem test sets connected to direct-access test connectors and other elements of the STC.

Tests to be Conducted--

1) System test;
2) Parameter variation tests.

This sample configuration is shown in Figure 3.1-8. In this configuration, there are three primary data-accessing paths and three control modes. The data accessing paths are:

1) From the Spacecraft Subsystems, via subsystem direct access test connectors, to the SSTS. At this point, data flow to the CDCS via the STC/SSTS adapters and also to the subsystem test set processing circuitry. (Detailed description of the SSTS is contained in Section 4.0.)
2) From the Spacecraft subsystems, via an umbilical, to the CDCS. At this point, data flow to the CDCS data processing circuitry and, in parallel, to the appropriate subsystem test sets via the STC/SSTS adapters.
3) From the Spacecraft Subsystems, via an rf link, to the radio subsystem test set and then to the CDCS via the radio SSTS adapter.

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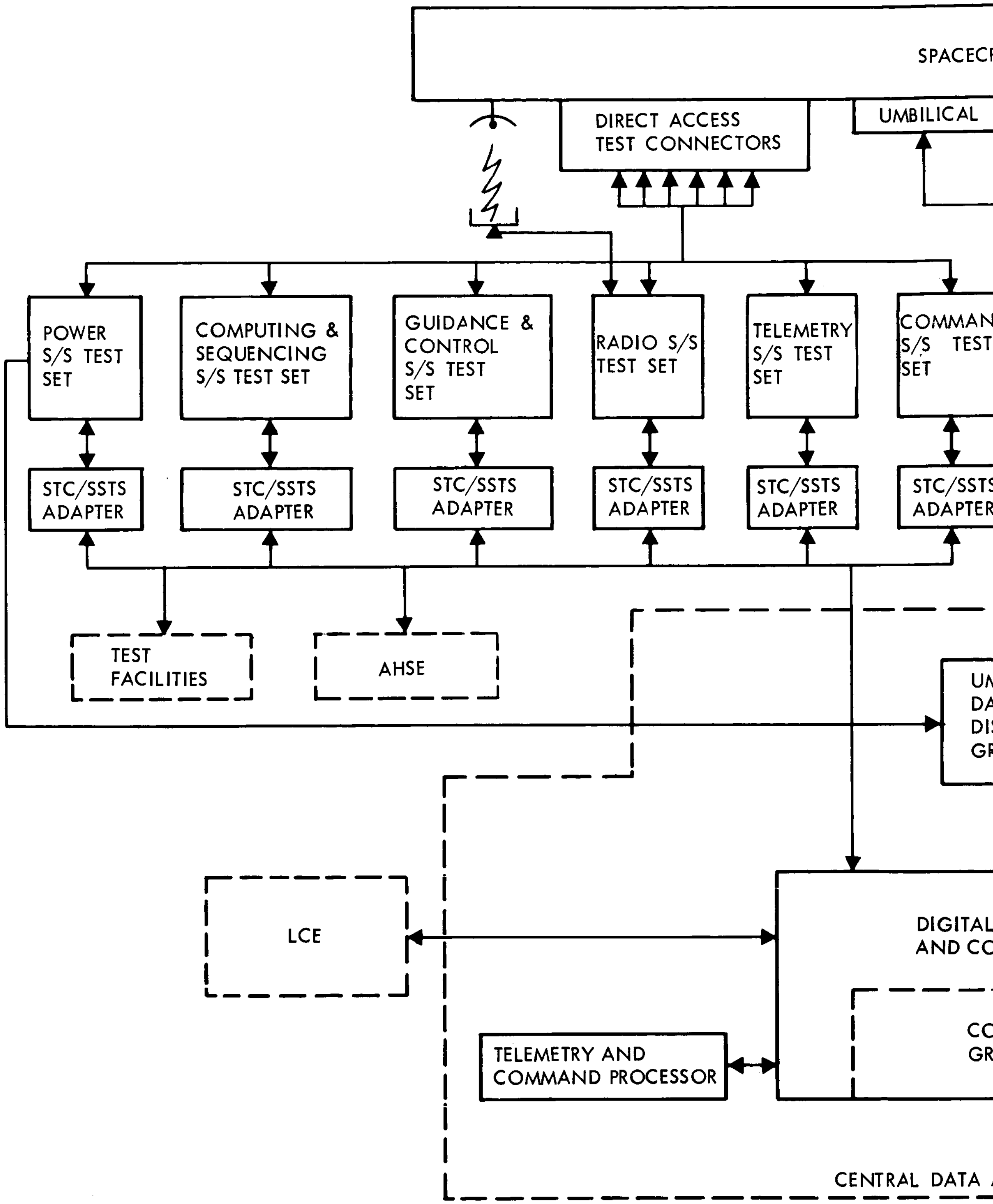

SPACECR
DIRECT ACCESS TEST CONNECTORS
UMBILICAL
POWER S/S TEST SET
COMPUTING & SEQUENCING S/S TEST SET
GUIDANCE & CONTROL S/S TEST SET
RADIO S/S TEST SET
TELEMETRY S/S TEST SET
COMMAN S/S TEST SET
STC/SSTS ADAPTER
STC/SSTS ADAPTER
STC/SSTS ADAPTER
STC/SSTS ADAPTER
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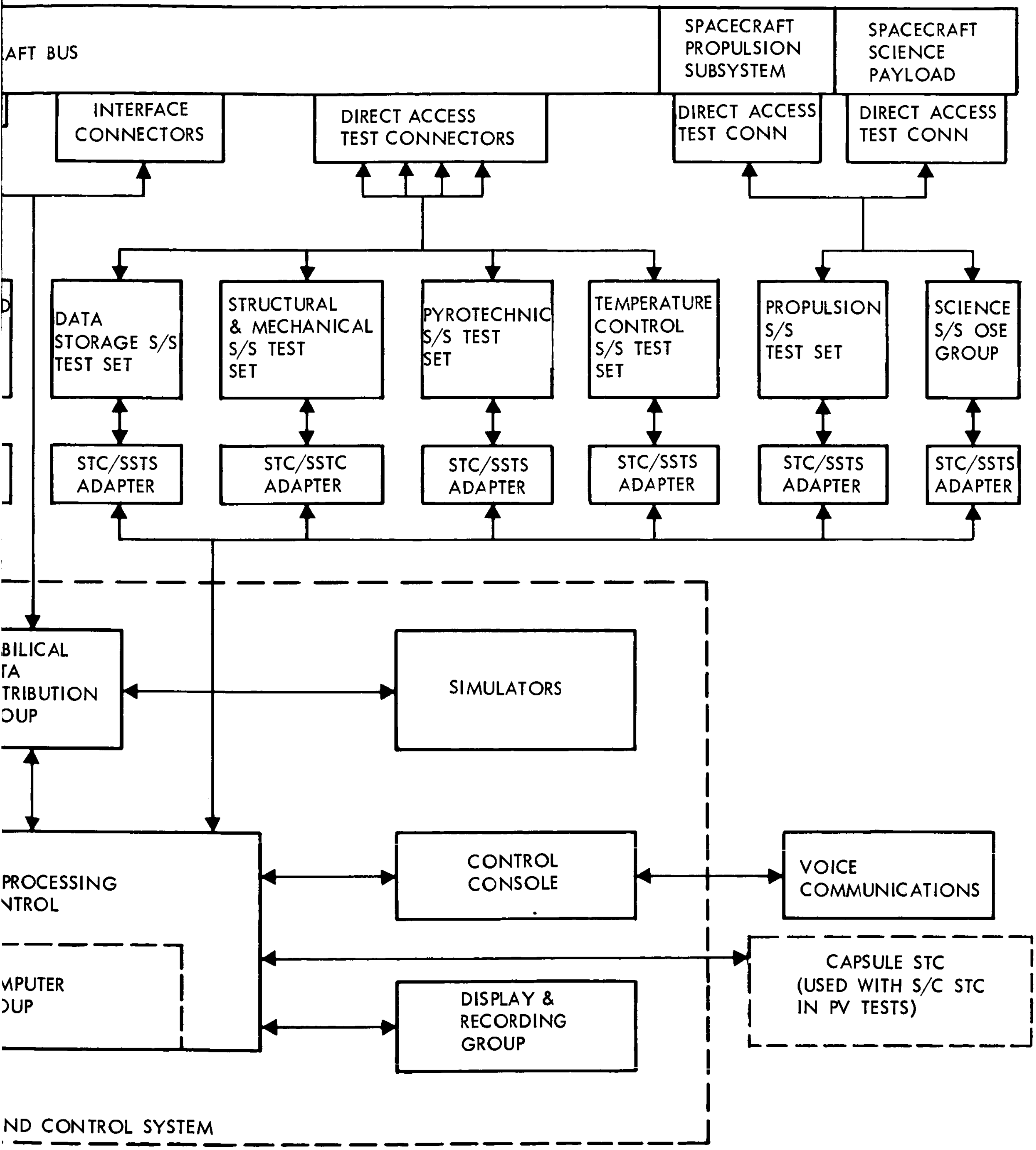

Figure 3.1-8: System Test Complex Block Diagram

The data form in each of these routes is significantly different and requires different processing.

The first route (direct access) provides access to data in the same form as existed in all previous tests in which the SSTS have direct access to the subsystems. Since the SSTS must be able to test their subsystems independently of the other STC elements, they have, inherent in their designs, the capability to process and evaluate these data. To provide central recording, monitoring, and evaluation, the CDCS processes and records these data in parallel. The STC/SSTS adapters provide signal conditioning and conversion of these raw data to permit CDCS processing and provide isolation of the parallel loading effects of the CDCS from the direct-access data lines.

The second route (the umbilical) handles a different form of data in that most of the analog data, available via the direct-access test connectors, is absent. The primary data form here is the telemetry stream, which contains engineering and scientific data from the Science Subsystem and engineering data from the Spacecraft Bus in the form of digital data riding on subcarriers. (There are, however, certain discrete and analog signals available on the umbilical that are routed separately to the appropriate SSTS via the CDCS junction box and processed as described in the first, direct access, route.

The CDCS receives these data and records them for later processing at other locations, such as the SFOF. In parallel with this recording, the CDCS demodulates, decodes, and separates these data into the Science Subsystem data and the Spacecraft Bus data. The science data are sent to the Science Subsystem OSE without further processing. The Spacecraft

Bus data are further processed within the CDCS so that calibrated engineering units of measurement are available in a serial digital form for transmission to the STC/SSTS adapters. At the SSTS adapters, the SSTS operators may select the measurements to be displayed.

During the CDCS processing of the Spacecraft Bus data, discussed above, secondary functions occur. First, a recording is made of the data sent to the STC/SSTS adapters; second, the data are placed in the CDCS computer memory to enable real-time plotting and machine monitoring. These functions are discussed in greater detail in Section 3.1.5.2.

The third data route (rf link) is from the spacecraft antenna to the CDCS through the radio subsystem test set receiving antenna, receiver, and demodulator. The output of the demodulator provides data to its own STC/SSTS adapter in the same form (digital data riding on subcarriers) as the telemetry data on the umbilical lines. The STC/SSTS adapters send these data to the CDCS where they are routed and processed as described above for the second data route.

Three control modes are available in this representative configuration: central preprogrammed mode, central manual mode, and local mode.

In the first mode (central preprogrammed) the CDCS computer sends sequential commands to the tape programmers within the SSTS via the STC/SSTS adapters. The SSTS then execute a controlled system test by executing selected, preprogrammed subsystem test routines.

In the second mode (central manual) the test director can command the subsystem test sets, either through the computer or via the voice communication system, to perform a system test on a step-by-step basis.

In the third mode (local) the control shifts to the subsystem test sets, each of which performs a selected portion of the system test in accordance with previously documented procedures. The primary roles played by the CDCS in this mode are: monitoring, recording, and providing alarms.

3.1.5.2 Major STC elements--Configurations and Functional Descriptions

This section describes the configurations and functional operation of the STC major elements identified in Section 3.1.5.1.

Descriptions of the internal and external STC functional interfaces are included. Environmental control and power and cabling systems are also described.

Subsystem Test Sets (SSTS)--The SSTS are those elements of Spacecraft Bus, Propulsion, and Science Subsystem OSE used in the STC. The specific SS OSE elements used for STC operations are identified and described in Section 4.0. The SSTS are used in the STC for two primary purposes:

1) To provide individual subsystem direct-access stimuli and evaluation capabilities under coordinated control at the CDCS.
2) To provide individual subsystem test, fault isolation, and analysis capabilities independent of the rest of the STC.

The SSTS have two basic interfaces:

1) Direct-access test connections to the Spacecraft Subsystems. This interface allows stimuli inputs to the Spacecraft Subsystems and accessing of test data from the Spacecraft Subsystems. The

cabling providing this interface is the same for all STC test areas as well as for subsystem test.

2) STC/SSTS Adapter Interface. This interface allows for data and control signal flow to and from the CDCS. Direct-access test connector data are sent from the SSTS to signal conditioners in the STC/SSTS adapters where they are converted and routed to the CDCS. Umbilical connector data, accessed by the CDCS, are sent to the SSTS via the STC/SSTS adapters. Test control signals from the CDCS are sent to the SSTS programming circuits via the STC/SSTS adapters. SSTS alarm circuits are interfaced with the STC alarm system via the STC/SSTS adapters.

In addition to the above, certain SSTS have unique operations with respect to the STC.

1) Radio SSTS. The radio SSTS is used to accomplish the rf interface with the spacecraft. It receives and routes rf telemetry data to the CDCS or the telemetry SSTS. Rf transmission of commands generated by the CDCS or the command SSTS is provided by the radio SSTS.
2) Command SSTS. Generates spacecraft commands and sends them to the spacecraft via the umbilical or the radio SSTS.
3) Science Subsystem OSE. Receives stripped-out serial telemetry data from the CDCS.

STC/Subsystem Test Set Adapters (STC/SSTS Adapters)--STC/SSTS adapters are provided for each Spacecraft Bus SSTS, and the spacecraft science and propulsion SSTS.

The STC/SSTS adapters, depicted in Figure 3.1-9, serve five primary functions:

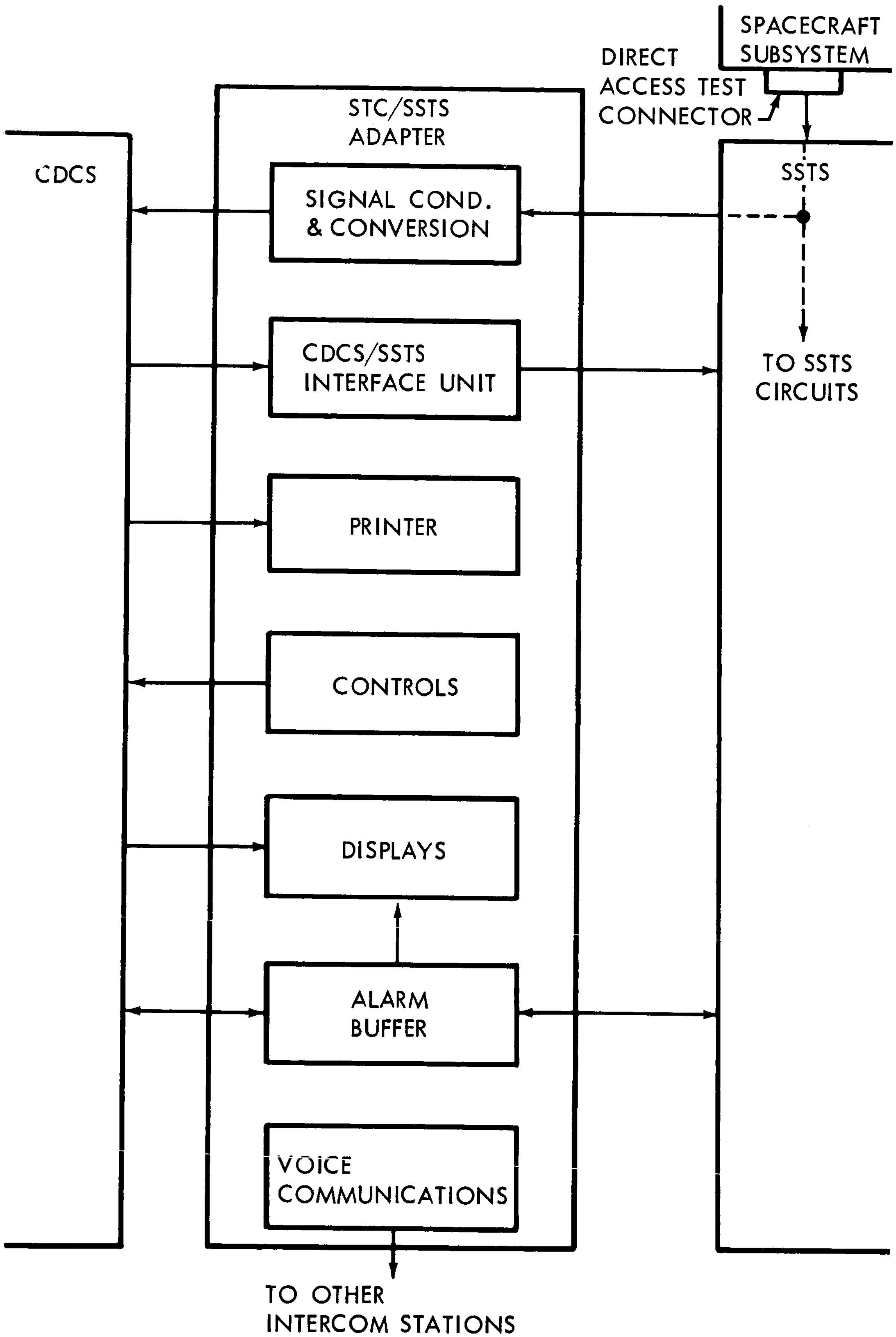

Figure 3.1-9: STC/SSTS Adapter Block Diagram

1) Provide CDCS/SSTS control;

2) Provide CDCS accessing of subsystem test data;

3) Provide display of CDCS processed data;

4) Provide display of alarm and status data;

5) Provide voice communication.

These functions are provided by the following equipment within each adapter:

1) CDCS/SSTS interface unit, containing a command converter for converting CDCS commands to SSTS commands;

2) Signal conditioning and sampling unit, containing signal conditioning and sampling modules for accessing, conditioning, and sampling spacecraft subsystem test data for delivery to the data converter unit.

3) Data converter unit, containing analog-to-digital and parallel-to-serial converters for formatting data for delivery to the CDCS;

4) Telemetry display and control unit, containing data converting, storage,and display modules and front-panel control for selecting and displaying any six analog and any six discrete data channels available on the CDCS-supplied telemetry serial pulse data train (it also contains control for the character printer.);

5) Character printer, which prints out hard copies, on command, of the displayed telemetry data and time and test references;

6) Alarm and status display unit, containing elements for visual alarm displays, alphanumeric display of test status, Greenwich Mean Time, spacecraft time, launch status, and countdown time;

7) Closed-circuit television (CCTV) repeater display unit, containing an 8-inch cathode ray tube (CRT), power supply, and associated electronics; it displays selected test data in parallel with the test director's CCTV;

8) Voice communications unit, containing a front-panel headset interface and voice-channel selector.

Central Data and Control System (CDCS)--The CDCS provides for integrated central control and data processing during system-level testing. It is also employed during the launch phase to condition, evaluate, and control the spacecraft and to supply spacecraft status to the launch control center. A functional diagram of the CDCS is shown in Figure 3.1-10.

The CDCS is composed of:

1) Test director's display and control group;
2) Alarm and monitoring subsystem;
3) Voice communications subsystem;
4) Umbilical data distribution subsystem;
5) Computer subsystem;
6) Telemetry and command processing subsystem.

Test Director's Display and Control Group--The display and control group provides the test director with the necessary display and control elements to allow him to exercise overall test responsibility to select test modes, select test sequences, monitor system testing and status, and direct fault isolation routines. Isometrics of the display and control group are described in Section 3.1.8 and shown in Figure 3.1-13 (Page 3-57).

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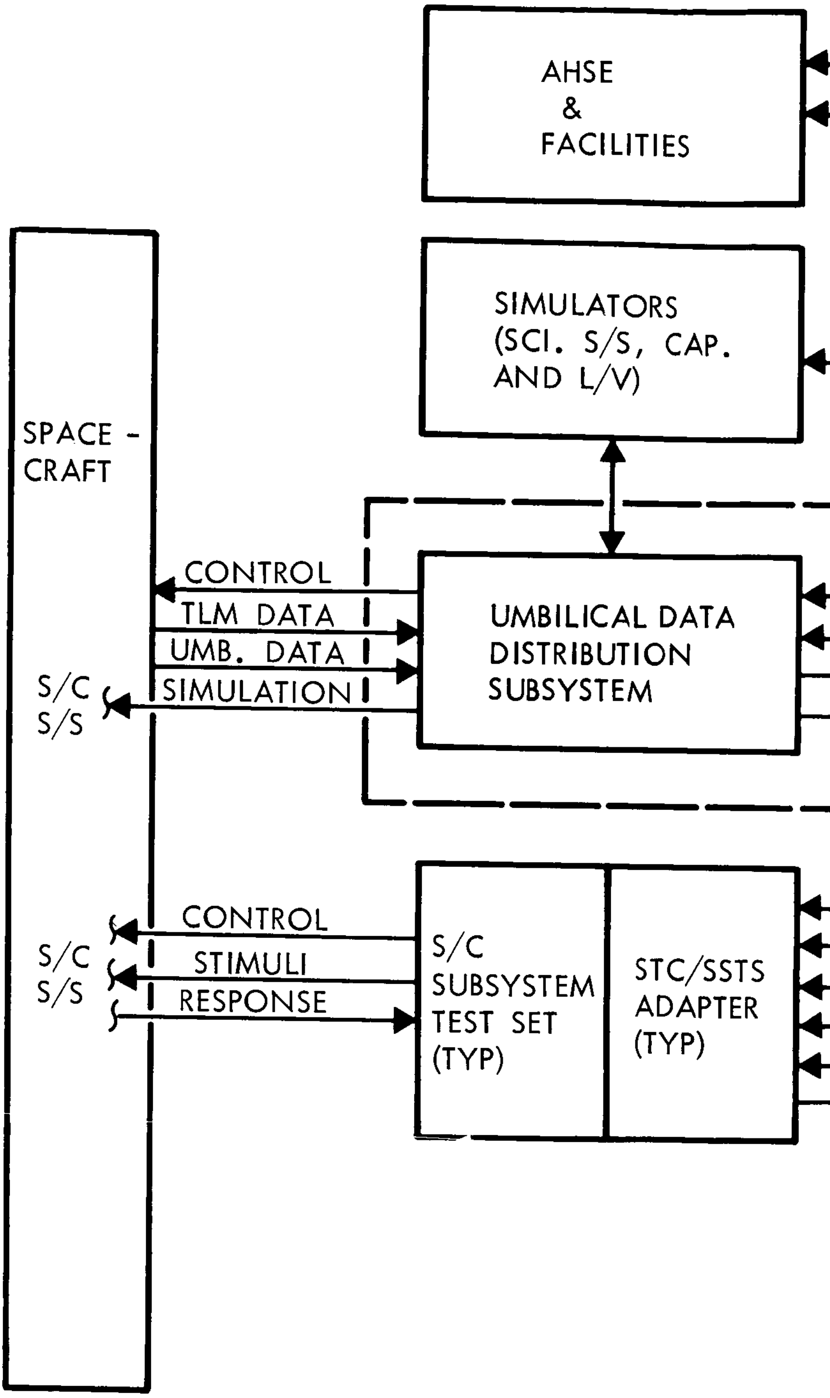
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&
FACILITIES
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(SCI. S/S, CAP.
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SIMULATION
S/C
S/S
UMBILICAL DATA
DISTRIBUTION
SUBSYSTEM
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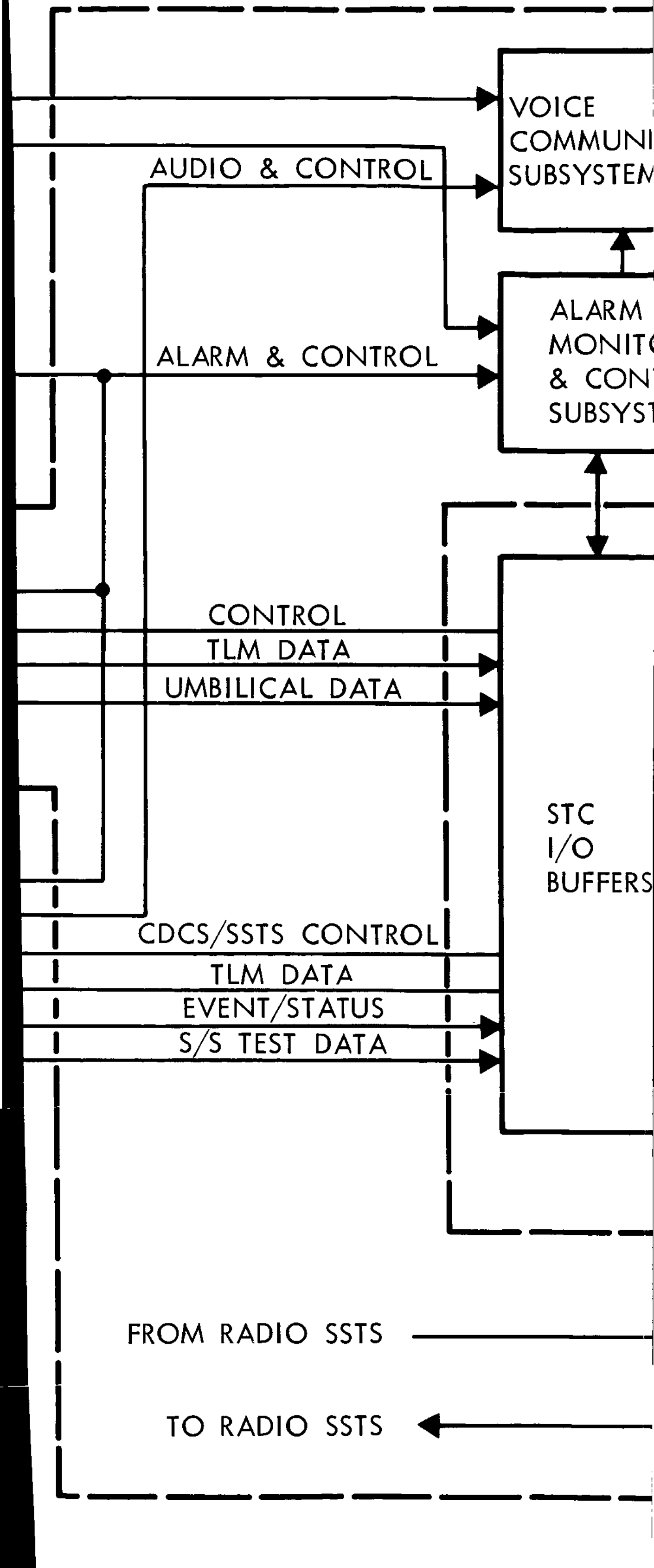
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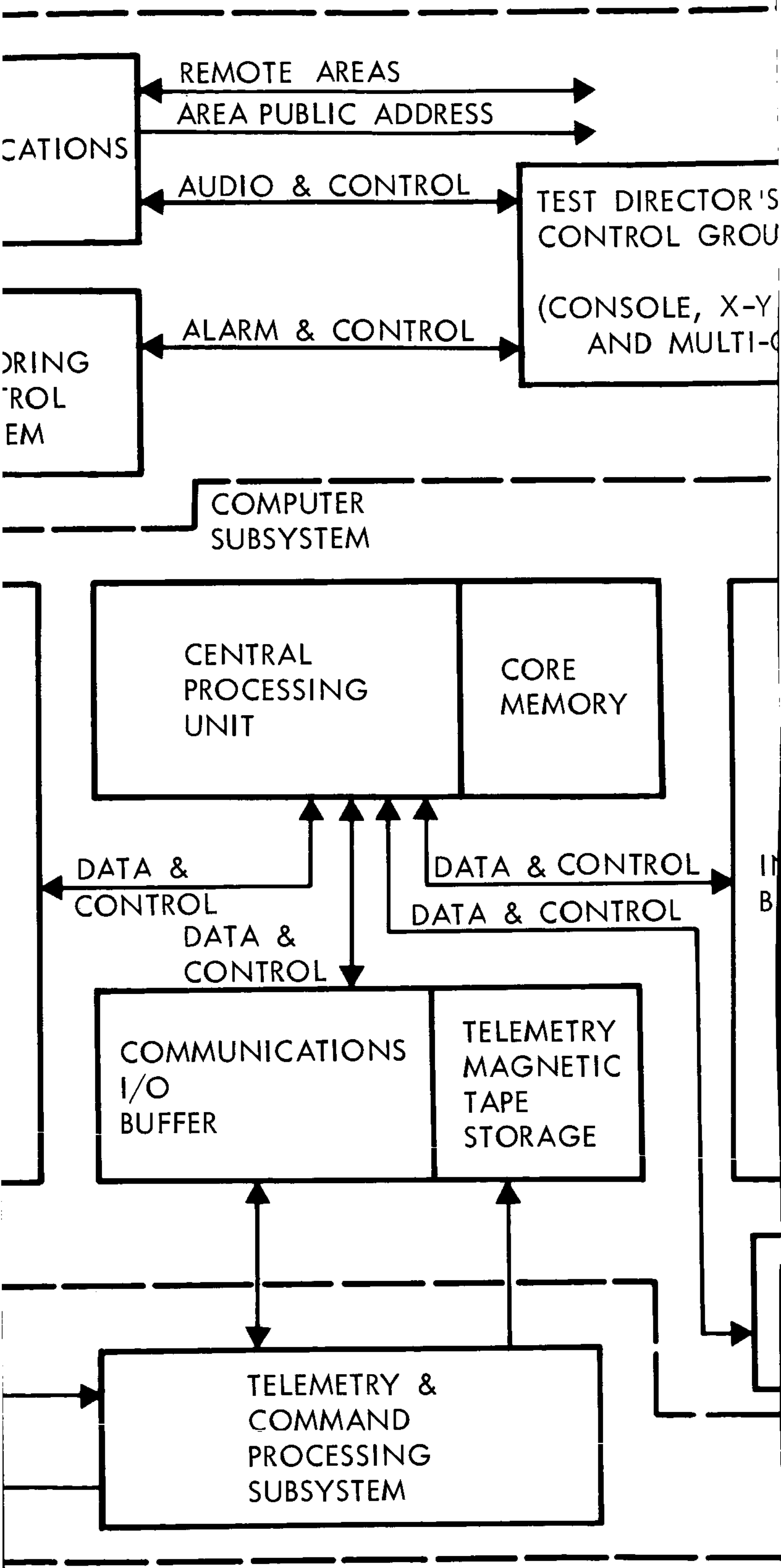

Figure 3.1-'10: Central D

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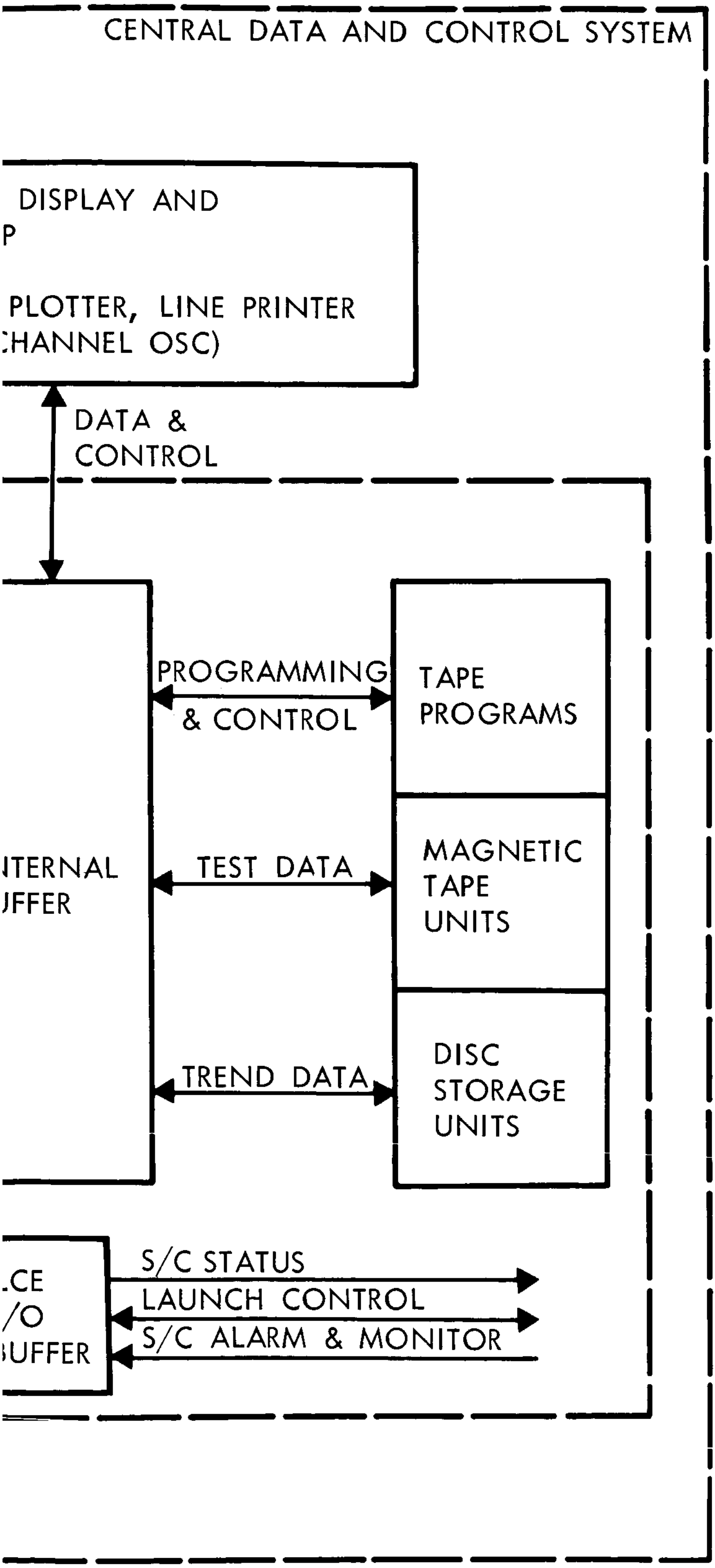


ta And Control System Data Flow Diagram

4

The display and control group consists of:

1) The test director's display and control console;

2) X-Y plotter;

3) High-speed line printer;

4) Multichannel oscillograph.

- Display and Control Console--The console provides control of system-level testing in three primary modes: central, manual, and local. In the central mode, control is exercised by activating test selectors on the console to command the initiation of prepared test programs stored in the central data system. The manual mode allows the director to assume control of computer programs. Through typewriter input, he can select specific tests or groups of tests that are part of the stored program. The local mode allows test authority to be transferred to the subsystem test sets.

  The director is also able to access and display real-time and stored test data through use of typewriter input to executive routines. The data may be displayed in alphanumeric and pictorial form. Alphanumeric and legend, event, and status display are provided to give the director current and immediate reference as to the test status of each major item of the STC. A CCTV monitor is provided to give the director area-surveillance information such as TV camera views of the spacecraft test area. Other camera views, such as one of the X-Y plotter, may be switched in to provide the director with real-time data plots. The console also provides for voice communication to all local and remote STC areas, area public address, and external telephone connections.

- X-Y Plotter--A floor-mounted X-Y plotter located near the director's console is provided to allow hard-copy data and mission profile plots to be made during system-level testing and mission simulations.

- High-Speed Line Printer--The multiple hard-copy line printer located near the director's console is the main test log. During each test routine, the computer-driven printer will print out data identifying the test being conducted, spacecraft equipment under test, STC and facility equipment identification and status, as well as significant events and time relationships and test results. Additional entries into the test log may be made through the console typewriter.

- Multichannel Oscillograph--A 20-channel oscillograph located near the test director's console for recording up to 20 channels of analog data is available on command from the CDS.

Alarm Monitoring and Control Subsystem--The alarm monitoring and control subsystem interfaces with the CDS, test director's console, and major STC equipment items The system consists of independent critical-parameter sensors, safety interlocks, and control lines. In the event of a critical or hazardous condition, the system will trigger visual and audible alarms and activate the stored fail-safe computer routine. Display of and controls to safe the spacecraft elements are provided on the console. In the event of main power failure, these functions are powered by the emergency battery supply.

Voice Communications Subsystem--The voice communications subsystem provides voice communication between the test director and all manned stations of STC, area public address, and telephone communication to remote test areas. The test director has priority-interrupt over all communications between stations in the STC and remote areas. In addition, in the event of main facility power failure, a portion of the voice network is powered by the emergency battery supply.

Umbilical Data Distribution Subsystem--This subsystem provides the collection and distribution interface for the spacecraft umbilical and simulators. All connections are made through a junction box. The J-box accepts and distributes umbilical data to the subsystem test sets and provides the connection interface between the spacecraft and simulators. In addition, the J-box contains signal conditioning and sampling units and a data converter unit for conditioning, sampling, and formatting spacecraft umbilical and simulator data for the CDS. Spacecraft umbilical hardline telemetry is fed through the J-box to the CDS.

Computer Subsystem--The computer subsystem consists of: general-purpose computer; data processing and control unit; magnetic-core storage units; disc storage units; magnetic-tape storage units; internal buffering unit; communications input-output buffer; STC input-output buffers; LCE input-output buffer; computer software.

The computer subsystem provides five basic functions during STC testing: (1) central test control, (2) data subsystem control, (3) on-line data reduction, (4) off-line data reduction, and (5) on-line self-check. In addition, the same functions are provided during prelaunch operations and Functions 3, 4, and 5 are available during postlaunch phases.

- Central Test Control--For system testing, the primary control is vested in the executive control routine of the computer. The basic operation of this routine and its interaction with the components of the computer subsystem in performing the required function is discussed below. The computer subsystem interfaces are shown in Figure 3.1-10.

The test program routine provides the stimuli control signals via test sets or command via the radio command links for each test sequence. The taped test sequence is controlled by the executive control routine. The subsystem simulation functions required during subsystem testing are replaced by stimuli to the individual subsystems. The information flow is from the tape unit to the individual subsystem test sets via the computer and the STC input-output buffer and during launch operations via the communications input-output buffer, or LCE input-output buffer.

The individual subsystem test-set programmer-sequencers, command and control decoders, and buffer units use language compatible with the computer subsystem. Hence, the interface problems are simplified and test programs written for subsystem testing are employed for system testing. In addition, since the concept requires parallel-driven displays, standard units are employed from subsystem testing through launch.

The executive control routine monitors the alarm units associated with each subsystem. The function is performed by the interchange of data from each alarm monitor via the subsystem test-set buffer and STC buffer units. If an alarm occurs, the executive routine transfers control automatically to a fail-safe subroutine, by a priority-interrupt, and each subsystem is reset to a safe condition via their respective test sets. The subroutine also displays to the individual test set and the test director, upon command, the condition of the system at time of interrupt. The test sequences are initiated by the test director, who has three options:

1) Cancel test sequence;
2) Switch to semiautomatic mode to step the test program one step at a time;
3) Return to automatic test sequence.

- Data Subsystem Control--Under the control of the test monitor routine, data are received into the data processing and control unit via the input decommutators, STC input-output buffer, Communications input-output buffer, and LCE input-output buffer. The decommutated spacecraft and umbilical test data are addressed and distributed via the internal buffering unit to their respective magnetic-tape storage unit, disc file unit for trend data storage, and/or the fast-access magnetic-core storage unit.

The umbilical hardline PCM data are recorded in raw form and parallel processed for distribution via the STC input-output buffer to the various subsystem test-set adapters for display in engineering units.

The telemetry data are interfaced with the communications input-output buffer unit. The data flow is identical to the hardline for storage and real-time processing required to convert it to engineering units. After conversion, the telemetry data are also routed to the individual SSTE units for display and in parallel to the test director as required.

- On-Line Data Reduction--A limited amount of on-line data reduction is required. This is primarily associated with converting the basic telemetry data to engineering units, formatting the required command sequences, processing specific engineering data for the parallel displays, and formatting data for the high-speed line printer. The basic concept uses a set of parallel displays of critical parameters at the test director's console and subsystem test stations. These displays are driven by the computer subsystem employing a standard format. Routing is via the required input-output buffer under control of the test routine. Special display call-ups are required for the director's console. These are also supplied by the computer. The data from each test are stored on the multiple magnetic-tape units. Since the trend data are not required for on-line processing, and hence do not require fast access, they are routed to the disc file for storage.

- Off-Line Data Reduction--In off-line use, the computer facility can be employed for engineering analysis, test document generation, and trend analysis.

- Off-Line Self-Check--The computer subsystem may at any time be commanded into a programmed automatic self-check routine. In this mode, the computer subsystem initiates and status-monitors the self-check routines within each subsystem test set as described in Section 4.0, and concurrently self-checks its own elements, simulators, alarms and monitors, and support facility items. The validity of the self-check mode is ensured by scheduled equipment calibration and certification against secondary standards.

- Computer Software Programs--Descriptions of these programs are given in Section 3.6.4.2.

Telemetry and Command Processing Subsystem--The telemetry and command processing subsystem, performs all the functions necessary to receive and process raw telemetry data via the radio SSTS and to process and verify spacecraft commands for transmission by the radio SSTS. The equipment duplicates the command and telemetry processing functions of the DSIF and SFOF and uses an identical set of MDE. (MDE hardware is described in Section 3.5.) The use of MDE hardware in the STC ensures spacecraft, MDE hardware, and software compatibility.

<u>Simulators</u>--Simulators are used to supply the necessary electrical and mechanical interfaces for subsystem and system testing when the prime equipment is not available. The simulators provide essentially the same interface with the spacecraft as would the prime equipment, including equivalent electrical loads.

Figure 3.1-11 illustrates three simulators provided for STC use.

1) A launch vehicle system simulator is used in system test of the spacecraft at the contractor's facility and at AFETR prior to to assembly on the launch vehicle.

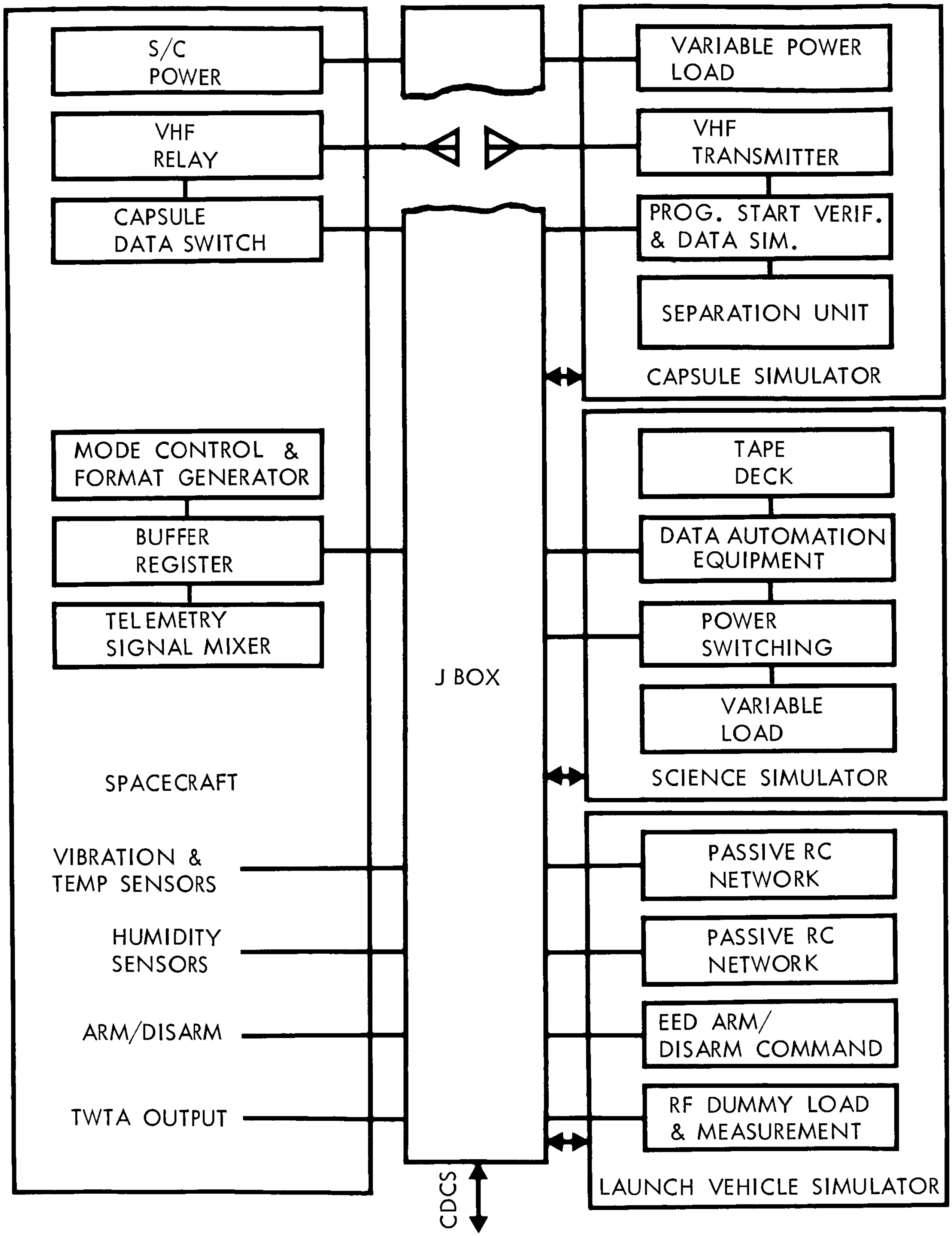


Figure 3.1-11: Simulators For STC Use

2) A capsule simulator is required in system test of the spacecraft when the actual capsule is not available.

3) A Science Subsystem simulator is provided in system test of the spacecraft when the actual Science Subsystem is not available.

<u>Support Facilities</u>

Environmental Control

- Spacecraft Environmental Control--The Spacecraft Bus is provided with cooled, dehumidified and filtered air from a mobile air conditioning unit. Control over the quality and quantity of the air supplied is a basic function of the air conditioning unit. A fail-safe monitor and alarm system interfaced with the CDCS is provided such that test operations can be stopped before damage to spacecraft equipment can result.

  Humidity, flow, and temperature of the incoming air, and temperature of the exhaust air, are monitored. If any of the four goes above a specified value, a visual and audible alarm is actuated at the test director's console.

- STC Environmental Control--The STC consoles in the CDCS will be supplied cooling air from a facility cooling air supply as required. The air will be routed to the appropriate consoles through ductwork installed underneath the false floor. The rest of the STC racks will be individually cooled, where required, by means of self-contained blowers.

Power and Cabling--See Sections 3.4.5.7 and 3.4.6.4.

- Spacecraft External Power--Spacecraft external power is provided by the power subsystem test set at all times during STC testing when the spacecraft power subsystem is off. This test set has an output capability of several a.c. and d.c. voltages as described in Section 4.2.1. These outputs to the spacecraft are controlled by the CDCS.

- Cabling--To facilitate installation, removal, or changes in the STC cabling, a modular raised floor is provided, with removable covers for ready access to the space between the raised floor and the building floor. All excess cable is strung in a random manner to reduce coupling and noise pickup.

  - Each cable contains a nominal 25 - 35% growth capability
  - Cable breakouts are not used - adequate junction flexibility is provided in the junction cabinets
  - All umbilical conductors are double-shielded
  - End covers are furnished for all connectors to provide for their protection when they are not in use
  - Cable ends and connectors are sealed and cushioned during transit to protect against an uncontrolled environment and rough handling

3.1.5.3 Support Equipment and Facilities Requirements

These requirements are described in Section 3.4.

3.1.6 Interfaces

STC-MOS--None. See Section 3.2.6 for LCE-MOS interfaces.

STC-TDS--None. See Section 3.2.6 for LCE-TDS interfaces.

STC-Spacecraft Bus--Hardware and test connectors, hardwire and umbilical connector, SS OSE via rf.

STC-Science Subsystem OSE--Direct access data, stripped telemetry, status and alarm, control, facilities power.

STC-Propulsion Subsystem OSE--Direct access data, status and alarm, control facilities power.

STC-Capsule OSE--Buffer register path, alarm, voice communications, facilities power.

STC-LCE--See Sections 3.2.5.1 and 3.2.6.

STC-STF--Voice communications, status and alarm, control.

STC-AHSE--Status and alarm, control.

STC-Facilities Power--Commercial telephone; space provisions; heating, cooling, ventilation; status and alarms.

3.1.7 Performance Parameters

The performance parameters of the STC less the SS OSE are identified below in six major categories.

3.1.7.1 Stimulation

Umbilical--Discrete commands and switching functions only. Other inputs are supplied by SS OSE.

External--Spacecraft sensor inputs (such as Sun and star) are provided by SS OSE. The STC provides electrical control of external stimuli, such as motion, temperature, and vacuum, which are generated by AHSE and test facility equipment.

3.1.7.2 Data Processing

1) Data accessing

a) PCM scanning system

(1) Inputs--1000 analog (800 spacecraft, 200 OSE) 800 discrete (600 spacecraft, 200 OSE)

(2) Sampling rate--All channels in 1 second

(3) High-speed sampling--50 channels, 5000 samples/second

(4) Accuracy--$\pm$ 0.1%

b) Serial data inputs (telemetry)

(1) Capacity--3 channels

(2) Data rate--50 kbps

c) Continuous access (alarm)

(1) Capacity--50 lines

2) Data Conversion--The CDCS performs the following types of data conversion:

a) PCM data values to engineering units

b) Identification coding of data values

c) Time-tagging of data values, both OSE and spacecraft timing

d) Data formatting for trend data analysis equipment

3.1.7.3 Evaluation

Four basic types of data evaluation are performed by the STC CDCS:

1) Event monitoring--Reacts to discrete event signals as they occur;

2) Data change--Reacts to change in data values;

3) Limit checking--Compares data values to stored program limits;

4) Arithmetic--Performs operations on related data and limit-checks result.

3.1.7.4 Display

The CDCS provides six types of data displays:

1) CCTV display system--A closed-circuit TV monitor display provides for area surveillance and camera views/of alphanumeric, vector plotting, and analog display of test data at the test director's console;
2) Digital (alphanumeric) display--Four displays are provided for test data and test parameters such as test time;
3) Status displays--100 indicators are provided for status and event change monitoring;
4) Alarms--Ten visual indicators coupled with one audio alarm system are provided;
5) Hardcopy printer--One line printer is provided for significant or demand data;
6) X-Y plotter--One X-Y plotter is provided with real-time plotting capability.

3.1.7.5 Recording

Four types of data recorders are provided by the CDCS:

1) Magnetic Tape--Four magnetic tape recorders are provided, each capable of recording raw PCM data at 50 kbps;
2) Analog recorder--20 channels of oscillograph recording are provided; also provides visual display.
3) Line printer--One (described above in Section 3.1.7.4).
4) X-Y plotter--One (described above in Section 3.1.7.4).

3.1.7.6 Control

Control Outputs--The CDCS has 20 digital control output channels for control of SS OSE, AHSE, and test facility equipment.

Control Inputs--The CDCS has four input devices each providing complete program control. They are a magnetic-tape reader, punched tape reader, an alphanumeric keyboard, and a card reader. In addition, manual switches are provided for specific control functions such as test hold, recycle, and test stop.

Control Interface Channels--Two control interface channels are provided, one for capsule STC and the other for LCE.

3.1.8 Physical Description - STC

The STC uses standard JPL racks as used on Mariner C plus standardized display and control consoles and specially packaged equipment such as an X-Y plotter and a line printer. An isometric view of an SSTS adapter is shown in Figure 3.1-12 and the test director's display and control console is shown in Figure 3.1-13. All equipment racks are mounted on a false floor with the STC cabling installed underneath. The equipment is arranged to provide the test director and all SSTS operators with direct viewing of the spacecraft. Table 3.1-3 lists basic STC equipment.

3.1.9 STC Reliability and Safety

The STC in its launch support configuration will be required to meet a specific numerical probability of success. Achieving specified reliability goals will be of prime importance during the design phase, where applicable redundancy, component derating, drift, and worst-case analysis techniques will be employed.

Safety of both personnel and the spacecraft have been considered in the STC design.

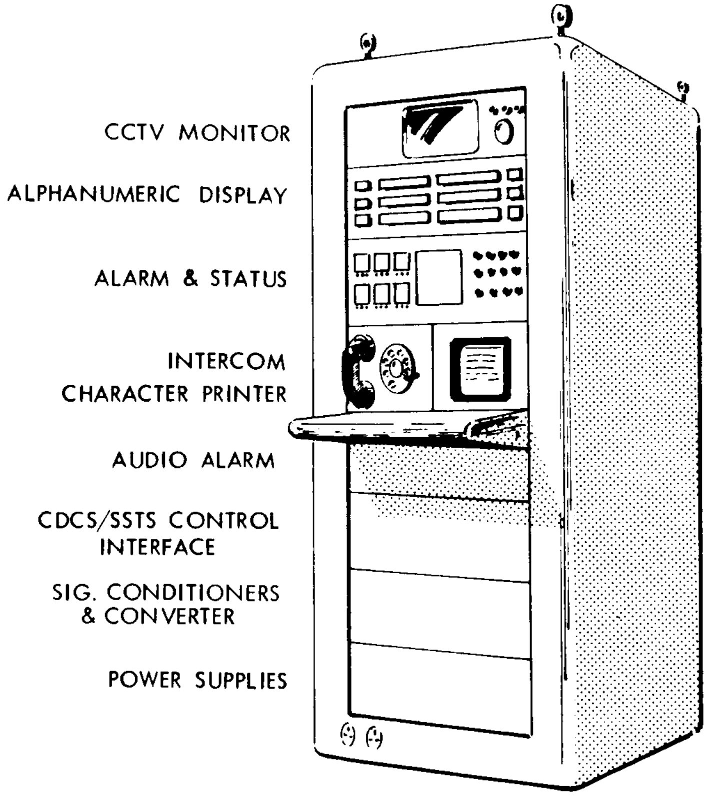


Figure 3.1-12 STC/SSTS Adapter

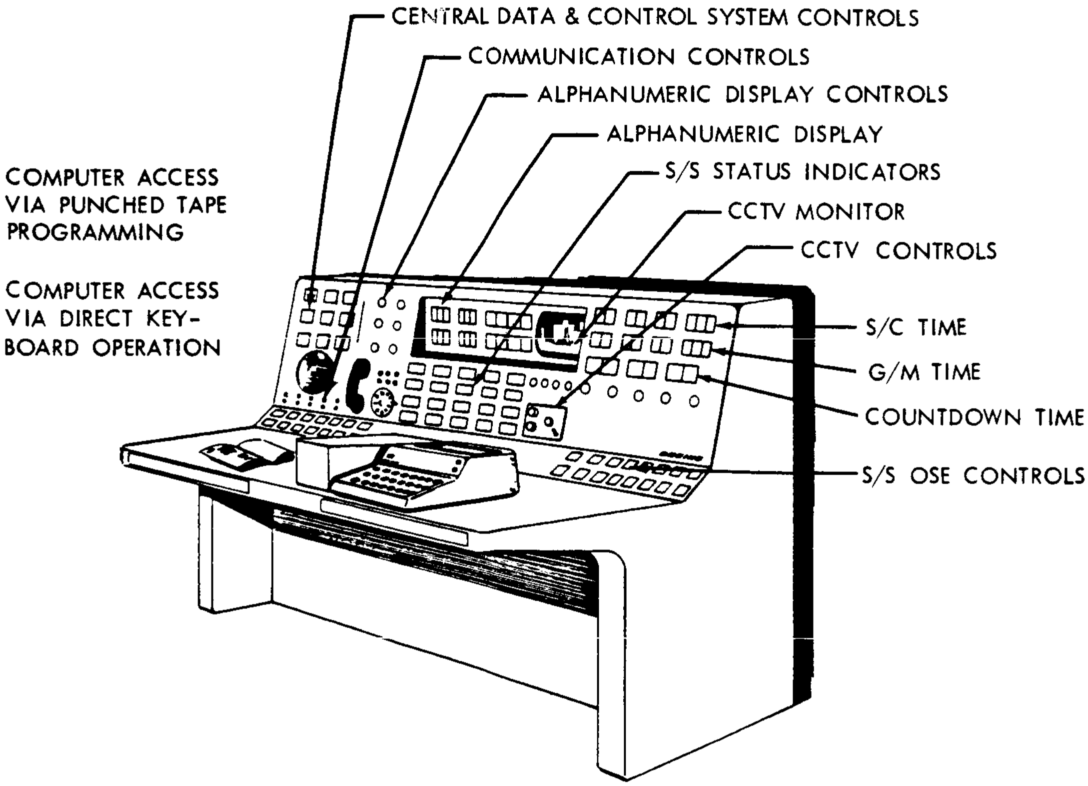


Figure 3.1-13: Test Director's Display & Control Console

Table 3.1-3: SYSTEM TEST COMPLEX EQUIPMENT

Central Data and Control System

- Umbilical data distribution group
- Digital processing and control group
- Display and control console
- Central display and recording group
- Computer group
- Telemetry and command processor

Simulators

- Launch vehicle simulator
- Science Subsystem simulator
- Flight capsule simulator

STC/SSTS Adapters

- Spacecraft Bus STC/SSTS adapter group
- Propulsion Subsystem STC/SSTS adapter
- Science Subsystem STC/SSTS adapter

Power and Cabling System

- STC signal cabling
- Alarm system cabling
- Power and ground cabling
- Power switching and distribution unit
- Emergency power units
- Intercom cabling

Voice Communications System

- Intercommunications system
- Public-address system
- Standard telecommunications

Subsystem Test Sets (SSTS)

- Spacecraft Bus Subsystem test sets
- Science Subsystem test sets
- Propulsion Subsystem test sets

1) A public address system and voice communications are provided at all test locations to alert personnel to dangerous situations and coordinate their actions.
2) Closed circuit TV is provided for area monitoring.
3) In the automatic testing mode, the computer program is designed to prevent damage from improper sequencing.
4) The alarm monitoring subsystem provides automatic detection and alarm for any abnormal spacecraft or test facility behavior.
5) A computer-stored fail-safe routine, activated by the alarm monitoring subsystem, shuts down the STC in the event of a critical or hazardous condition.
6) In the event of complete loss of main power, an emergency battery provides power for manual/remote control and indication of selected spacecraft functions.

3.1.10 Design, Development, and Test

The STC and LCE perform an important role in providing highly reliable Flight Spacecraft and maximum assurance of mission success. In the performance of spacecraft testing and checkout, delays due to test equipment troubles must be kept to a minimum. For this reason, a comprehensive design, development, and test program is planned.

The Design and Development Concept is based on the following items:

1) The use of off-the-shelf proven components wherever possible.
2) Where proven components are not available, instituting a thorough test program of existing components to obtain and document evidence that ample design margins exist for the intended application.

3) Undertaking new design of functional elements as a last resort, proving adequacy by:
   a) Building breadboards, using standard printed-circuit boards, modules, part derating, and best state-of-the-art concepts, and evaluate functional performance and margin limits;
   b) Build an engineering model (packaged) and test, fixing as required to get adequate margin;
   c) Document breadboard and engineering model configuration and test data, providing evidence of conservative design;
   d) Integrate with STC and LCE engineering test model (ETM) during evaluation and compatibility period.

Prior to the spacecraft type approval test, a test is performed on the STC using a spacecraft simulator to demonstrate and verify the readiness of the STC to support system testing. Following the verification test, an STC proof-test-model (PTM) ground integrity test is performed to verify that no unknown ground loops are present in the PTM (spacecraft) nor are any introduced when the STC is connected to the PTM. Verification of LCE compatibility with the PTM is accomplished by dummy-run tests.

3.2 LAUNCH COMPLEX EQUIPMENT

The primary functions of the Launch Complex Equipment (LCE) are to determine operational readiness of the Planetary Vehicles, to monitor status during final countdown, to control the spacecraft during final countdown, and to condition the spacecraft for launch.

3.2.1 Summary

The LCE is used after a Planetary Vehicle is enclosed within a nose fairing section, thus limiting access to that provided by the umbilical and the rf link.

Launch checkout of the Planetary Vehicles is first conducted in an explosive safe area, then at the launch complex. Figure 3.2-1 shows both configurations. The top configuration is used for checkout of the Planetary Vehicles, which are completely wired and shrouded, with either dummy or live propulsion charges installed as if for launch. The configuration at the bottom shows the same Planetary Vehicle installation and hookup mated to the launch vehicle on the mobile launcher.

Figure 3.2-2 depicts the locations of the major LCE items which participate in a launch countdown. Mounted on the mobile launcher (M/L) is equipment which conditions and routes spacecraft umbilical signals, provides external power for the spacecraft, and furnishes line-of-sight continuity for the rf link. RF signals are transmitted directly to the spacecraft checkout facility (SCF) under one mode of open-loop testing, and via DSIF 71 in the other mode. Umbilical signals, except for critical functions, are routed to the SCF for processing, evaluation and display; some selected information is then relayed to the blockhouse

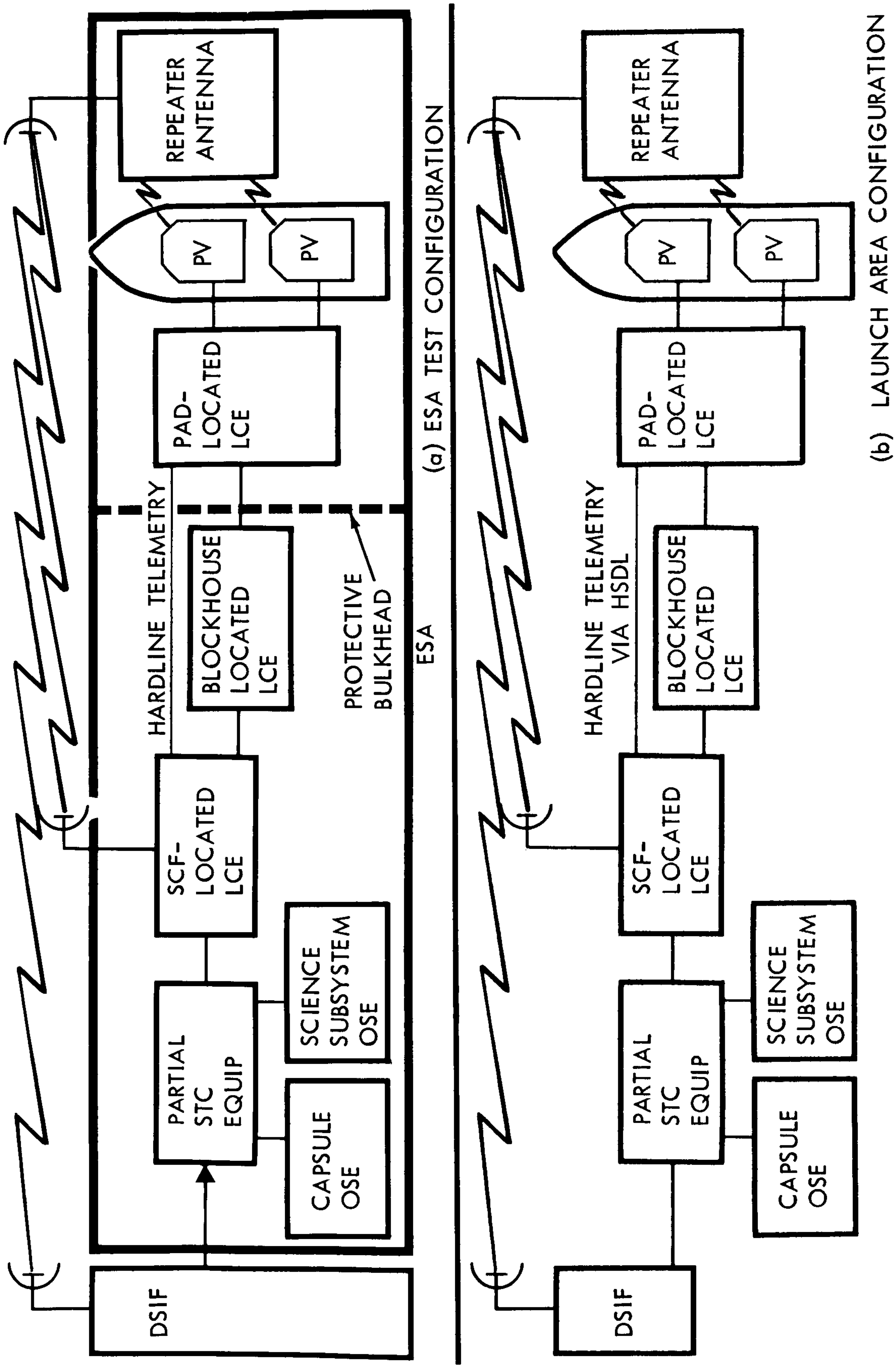
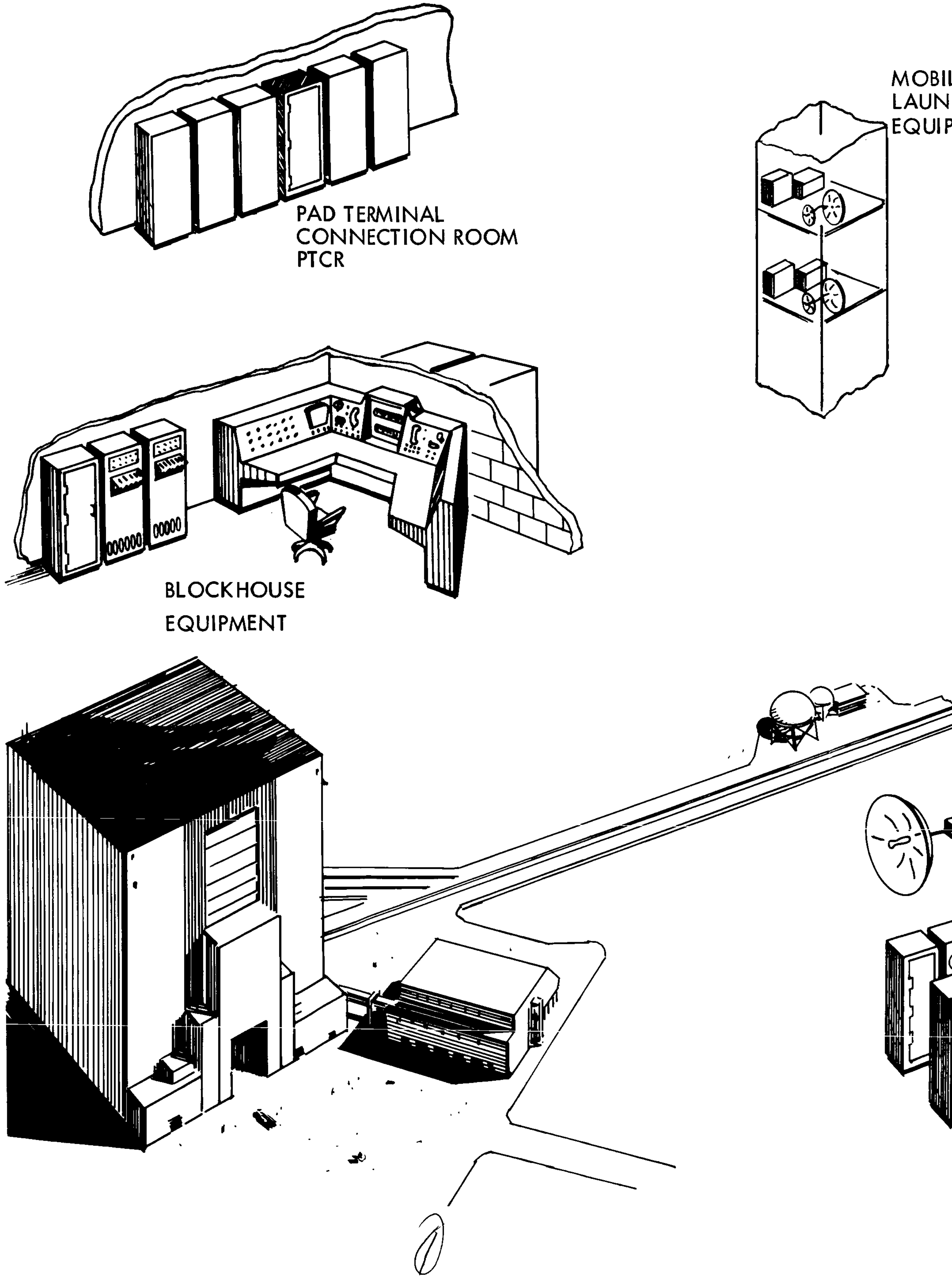


Figure 3.2-1: Major LCE Configurations


PAD TERMINAL
CONNECTION ROOM
PTCR
MOBIL
LAUN
EQUIP
BLOCKHOUSE
EQUIPMENT

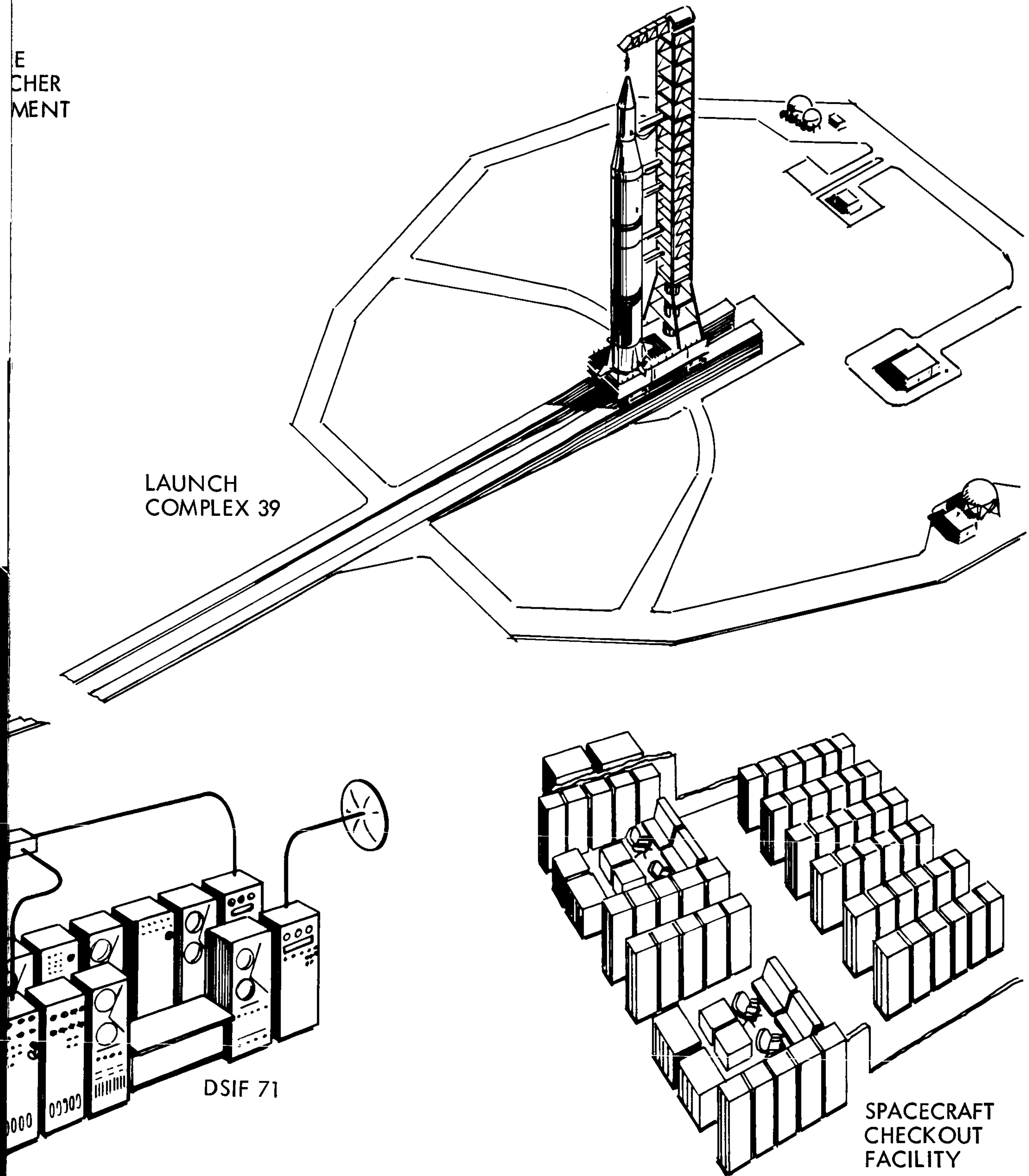


Figure 3.2-2: Principal Elements of the LCE

for display to the Planetary Vehicle test coordinator. Critical functions are continuously monitored and evaluated within the equipment mounted on the mobile launcher.

The LCE makes extensive use of the STC, augmented by signal conditioning equipment, and remote display, recording, and control equipment required because of separated geographic locations of equipment, spacecraft, and personnel. Many important advantages accrue from this use of the STC, such as increased confidence in mission success, increased equipment simplicity and reliability, decreased costs, and an improved test continuity which supports better data correlation and trend analysis.

This prominent use of the STC is due to the fact that it performs many of the functions required for launch checkout operations. Functions proposed for STC equipment include testing and manually controlling the spacecraft through the umbilical and rf link access; providing sufficient safeguards in the automatic mode to prevent damage due to improper sequencing, operator error or equipment malfunction; and providing interchangeability with respect to testing different spacecraft. A description of how these functions are performed is given in Section 3.1.5.

In performing the LCE functions, application of the STC is modified with respect to systems test and checkout operations in two ways: (1) the majority of the subsystem test sets are not used because the inputs from the test connectors will not be available, and (2) the programs included in the central data and control system are modified to include such functions as control of both spacecraft during final launch countdown and conditioning of the spacecraft for launch.

3.2.2 Applicable Documents and Drawings

Basic reference documents applicable to all Operational Support Equipment are listed in Section 2.1. Following is a list of documents applicable to LCE design.

1) Mariner C Operational Support Equipment, Spec. No. OSE/MC-1-110, dated June 14, 1963;
2) Mariner C Requirements for Spacecraft/Launch Vehicle Integration, Engineering Document No. 151, dated February 22, 1965;
3) Test and Operations Plan, Mariner C, dated October 30, 1963;
4) Dwg. J8902515 -- J-Box MC 209 JB1 Assembly (U/T LC OSE);
5) Dwg. J8901896 -- Envelope Drawing LC OSE Umbilical Tower Junction Box MC 209 JB1;
6) Dwg. J8902198 -- LC/OSE Master Block Diagram, Sht. 1 of 2;
7) Dwg. J8902198 -- LC/OSE Master Block Diagram, Sht. 2 of 2;
8) Dwg. J8902513 -- Wiring Diagram, J-Box MC 209 JB1, U/T LC OSE;
9) Dwg. J8901874 -- JPL Launch Control Pad 12 AMR "Hard Line" Cable Block Diagram;
10) Dwg. J890175 -- JPL Launch Control Pad 13 AMR "Hard Line" Cable Block Diagram.

3.2.3 Design Constraints and Requirements

The fundamental constraints and requirements used to derive the LCE design were those contained in the documentation which accompanied the Phase IA, Task B Request for Proposal. The technique used to ensure compliance with these constraints and requirements is described in Section 3.2.3.1; exceptions taken to the fundamental constraints and requirements are identified in Section 3.2.3.2; derived requirements are identified in Section 3.2.3.3.

3.2.3.1 Compliance

A review was conducted of the applicable documents listed in Section 3.2.2. The goals of this review were twofold:

1) To identify every statement in the mission description, performance and design requirements, and guidelines documentation which had a potential impact on the LCE, and determine the nature and extent of the impact and to functionally categorize the constraints and requirements thus identified.
2) To update Boeing's understanding of Ranger-Mariner OSE designs and techniques and to evaluate their possible applications to the Voyager System.

This study revealed that the LCE design approach must closely follow that of the STC, since many required LCE performance functions were identical to or similar to those of the STC. The requirements which were peculiar to the conduct of the launch countdown were separated and used to identify those functions which the LCE would perform because the STC did not. This technique served to enhance compliance with each requirement.

With the specific LCE functions identified, their incorporation into a design approach was straightforward. The approach adheres closely to that of Mariner and, as the design matures, use of specific Mariner equipment will receive full consideration.

3.2.3.2 Exceptions

None

3.2.3.3 Derived Requirements

The following LCE requirements have been developed through an analysis of the spacecraft design and a sequential functional analysis.

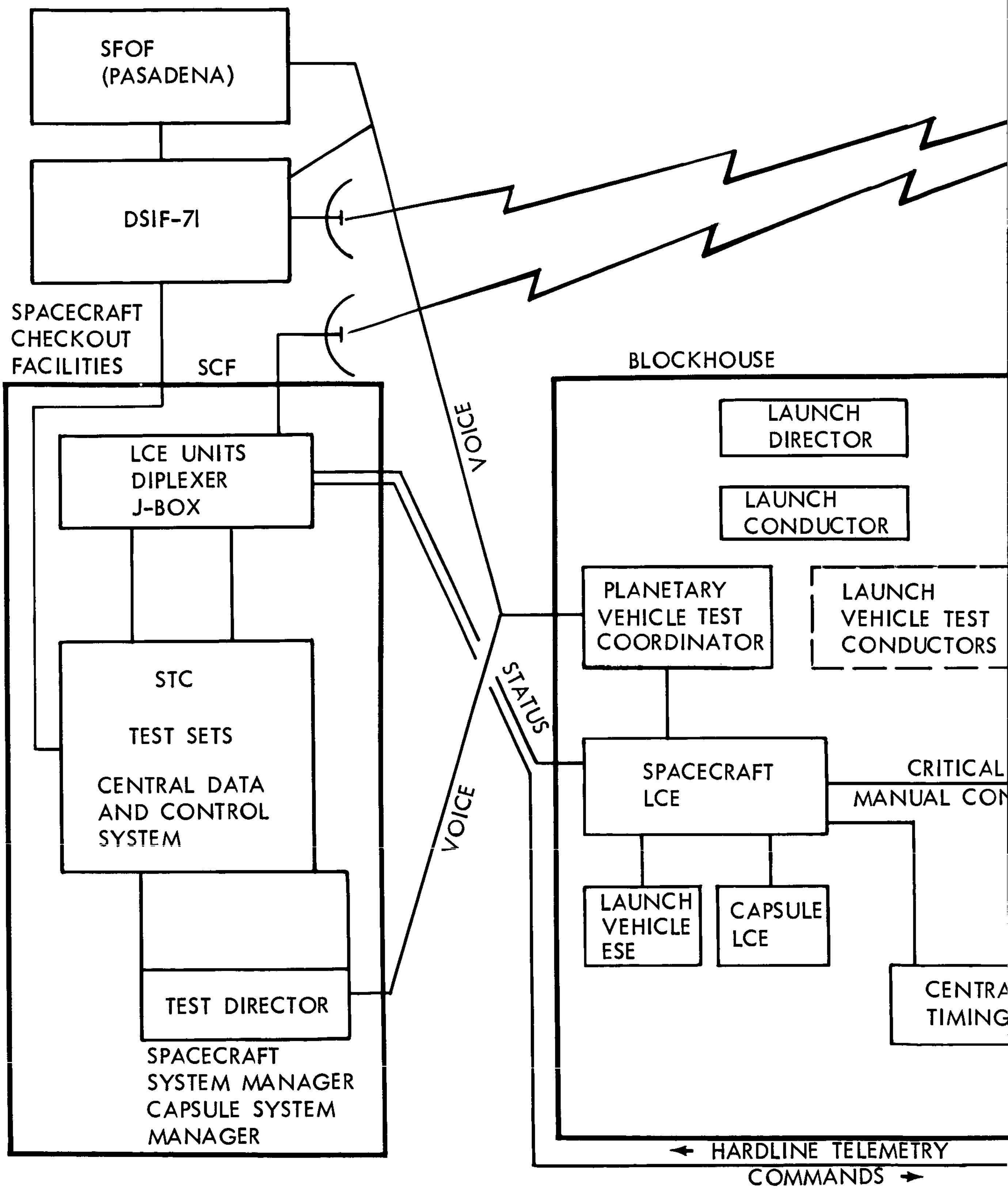
1) Two rf antennas and two umbilicals are provided which require simultaneous monitoring by the LCE;
2) Launch sequencing and T-count timing are provided by the Launch Vehicle System;
3) Emergency shutdown signals are provided to and from the Launch Vehicle System;
4) The subsystem specialists (part of the test teams) are located in the spacecraft checkout facility and provide technical consultation to the Planetary Vehicle test coordinator located in the blockhouse;
5) Various umbilical functions must be provided as described in Section 3.2.5.2.

3.2.4 Trade Studies

The principal trade studies conducted in developing the preferred design approach were carried out in the design of the STC. These studies are described in Section 3.1.4.

3.2.5 Functional Descriptions

Figure 3.2-3, which is an expansion of the launch area portion of Figure 3.2-1, contains information pertinent to the description of the LCE functions.


SFOF
(PASADENA)
DSIF-71
SPACECRAFT
CHECKOUT
FACILITIES
SCF
LCE UNITS
DIPLEXER
J-BOX
STC
TEST SETS
CENTRAL DATA
AND CONTROL
SYSTEM
TEST DIRECTOR
SPACECRAFT
SYSTEM MANAGER
CAPSULE SYSTEM
MANAGER
VOICE
STATUS
VOICE
BLOCKHOUSE
LAUNCH
DIRECTOR
LAUNCH
CONDUCTOR
PLANETARY
VEHICLE TEST
COORDINATOR
LAUNCH
VEHICLE TEST
CONDUCTORS
SPACECRAFT
LCE
CRITICAL
MANUAL CON
LAUNCH
VEHICLE
ESE
CAPSULE
LCE
CENTRA
TIMING
HARDLINE TELEMETRY
COMMANDS

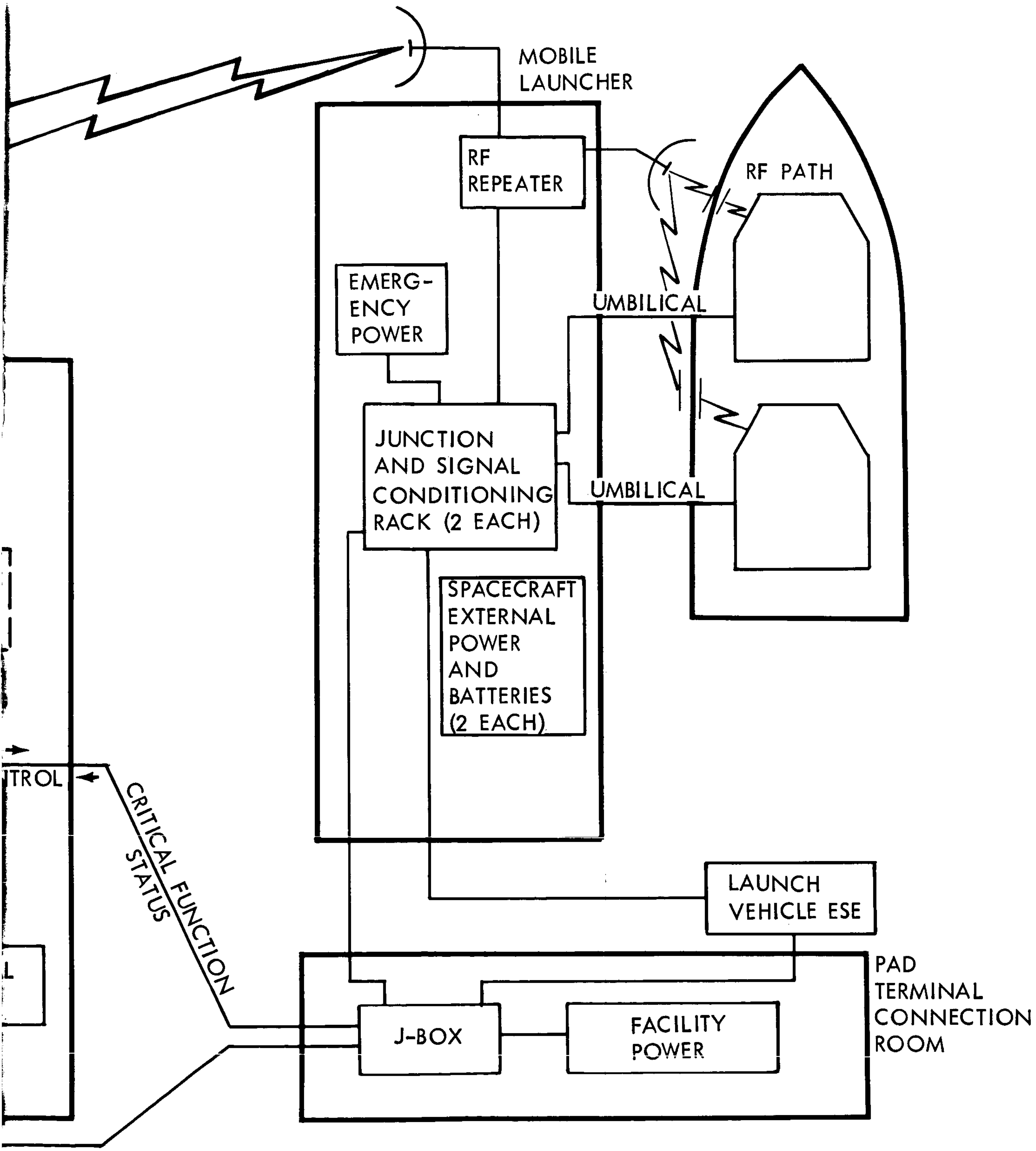


Figure 3.2-3: LCE Launch Area Configuration

2

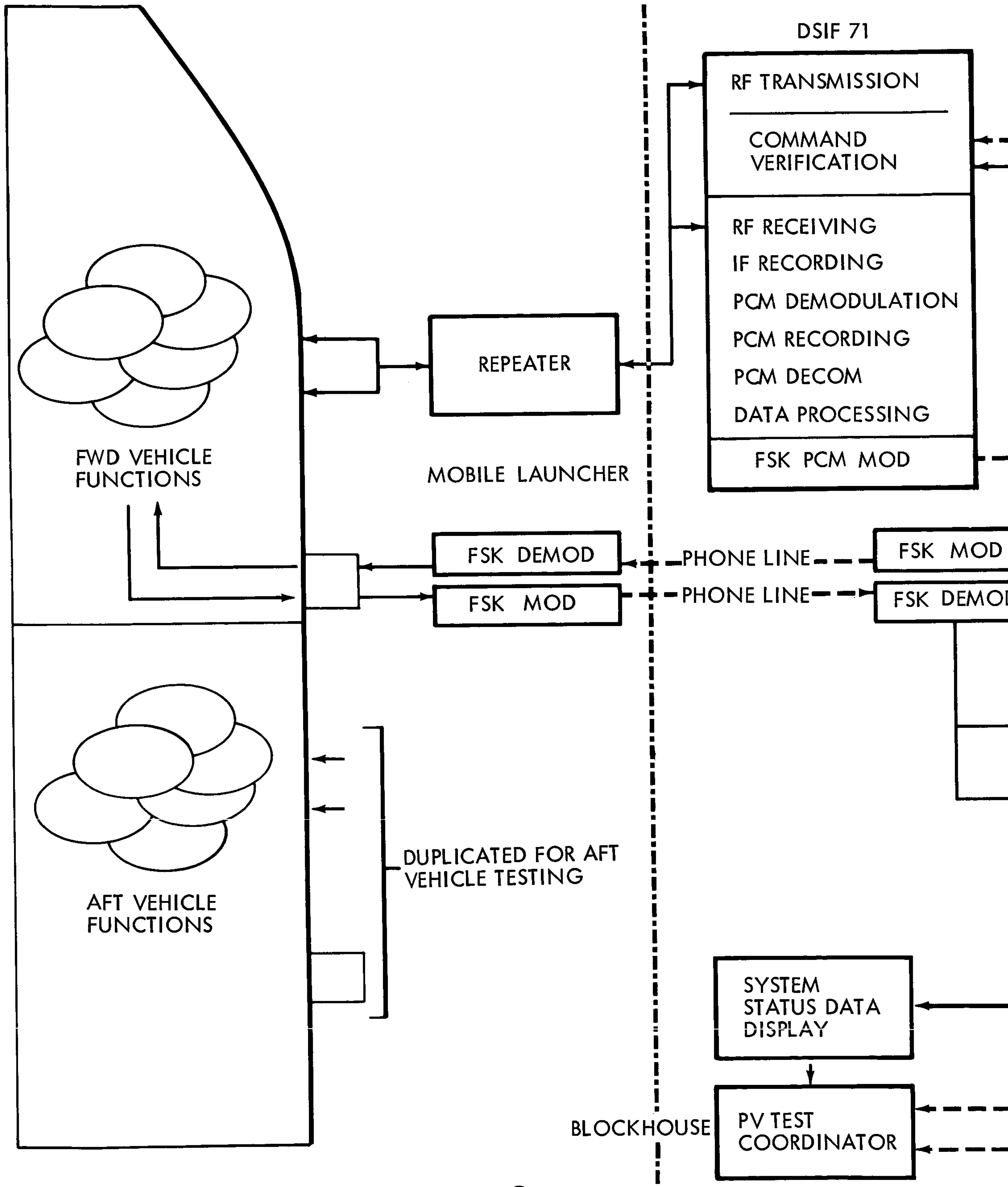
3.2.5.1 LCE Overview

The Launch Complex Equipment is used after the two Planetary Vehicles are enclosed in the nose fairing of the launch vehicle. Prior to this, testing is done by the spacecraft STC and the capsule test complex. The LCE makes maximum use of the STC, as brought out below.

General test configurations for the LCE are dictated by the testing requirements of two areas: the explosive safe area (ESA) and the launch area. The launch area configuration is treated here because it imposes the broader requirements for data handling and routing. However, the functional description is quite similar for both areas, the difference being mainly in the location of equipments. LCE information flow is shown in Figure 3.2-4.

Because the Planetary Vehicles are enclosed in the nose fairing, the LCE communicates with the vehicles only through the umbilicals and the rf link; facility is provided for rf transmission through the nose fairing. The flight telemetry and up-link commands flow through the rf link; for operation during rf silence, the umbilical duplicates the rf link capacity and provides considerable supplementary access for monitoring critical functions, supplying external power, and for controlling and checking various sequences during countdown. The termination of the umbilicals and their mating with the LCE occur at the junction and signal conditioning rack (JSCR), located on the mobile launcher adjacent to the umbilicals. The JSCR provides access to all Planetary Vehicle functions available through the umbilical and contains the signal conditioning required for signals being routed to other portions of the launch complex and to remote areas.

B L A N K


DSIF 71
RF TRANSMISSION
COMMAND VERIFICATION
RF RECEIVING
IF RECORDING
PCM DEMODULATION
PCM RECORDING
PCM DECOM
DATA PROCESSING
FSK PCM MOD
REPEATER
FWD VEHICLE FUNCTIONS
MOBILE LAUNCHER
FSK DEMOD
FSK MOD
PHONE LINE
PHONE LINE
FSK MOD
FSK DEMOD
DUPLICATED FOR AFT VEHICLE TESTING
AFT VEHICLE FUNCTIONS
SYSTEM STATUS DATA DISPLAY
BLOCKHOUSE
PV TEST COORDINATOR

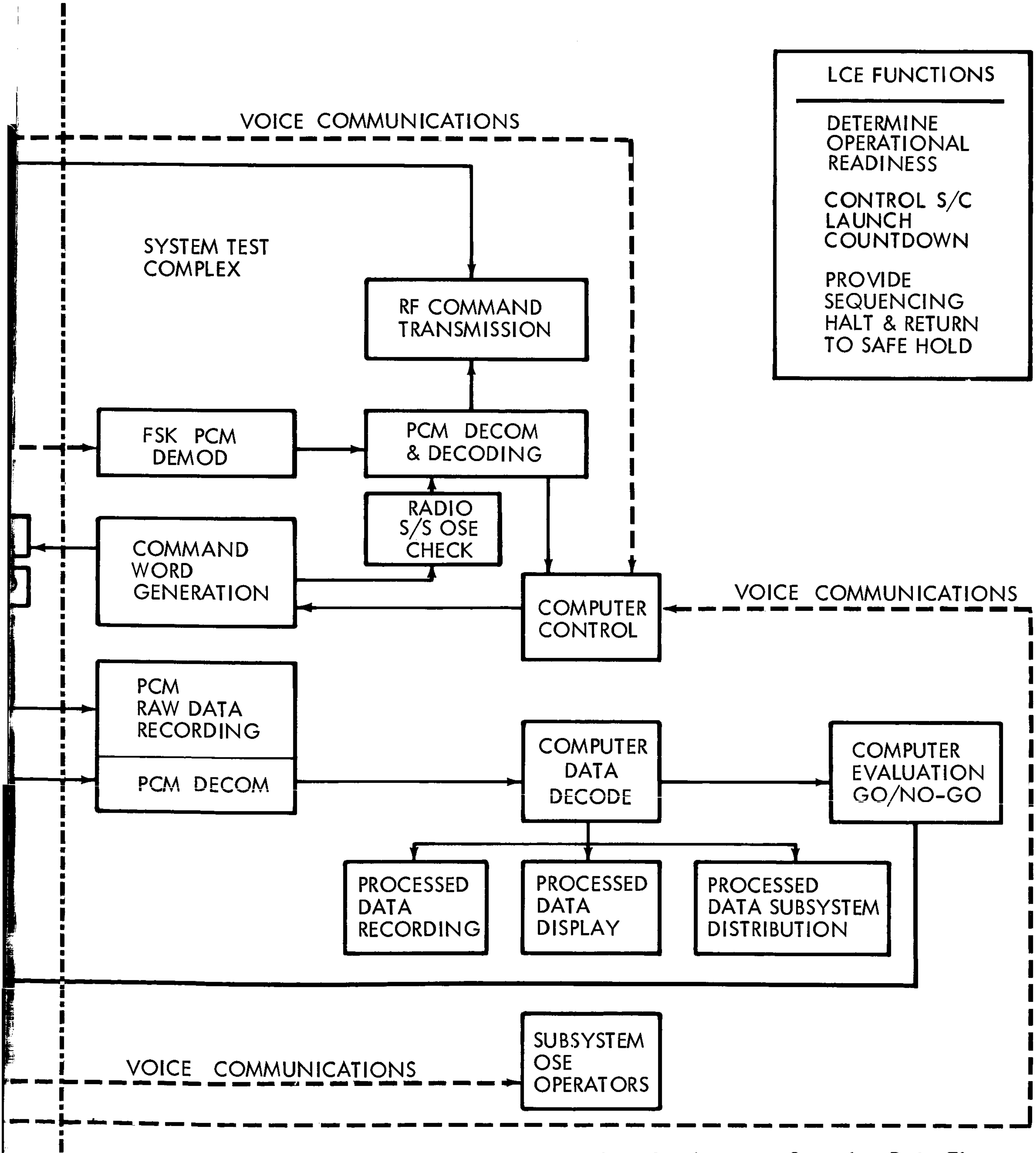


Figure 3.2-4: Launch Complex Data Flow

(7)

The rf link takes alternate routes from the Planetary Vehicles to the STC located in the spacecraft checkout facility; one route is through DSIF 71, the other is through the telemetry SSTS in the STC. The inputs from these two paths, together with a third TLM input to the STC through the umbilical, are recorded to accommodate subsequent comparison by use of the data-processing capabilities within the STC. This comparison is necessary in evaluating the compatibility and performance of the rf link via DSIF 71.

The up-link flow of commands from the STC to the Planetary Vehicle follows three similar routes, the difference being that, whereas the down-link telemetry stream is processed through the ground telemetry equipment, the up-link commands are processed through the command-generation and formatting equipment.

The STC, by virtue of its central data processing and control capabilities, performs many of the LCE functions associated with testing and conditioning the spacecraft for launch. Thus, detailed status information on the Planetary Vehicles and complete control of the spacecraft are provided to the Spacecraft System manager and capsule system manager, in the spacecraft checkout facility. Additionally, the STC extracts from the telemetry stream pertinent information for routing to the SS OSE and capsule OSE. Therefore, for the Planetary Vehicle, the spacecraft checkout facility is the center of data-processing operations. Control capability is available in the blockhouse to the Planetary Vehicle test coordinator. Since his primary function is to serve as a coordinator between the launch conductor and the overall spacecraft operations manager, his range of control need not be extensive. However, he must have complete status information on each Planetary Vehicle and all its subsystems. Detailed knowledge on individual subsystems is obtained from the STC through the use of a call-up capability.

As in STC operations, self-tests are performed before and after each launch checkout operation. Under control of the STC central data and control system (CDCS), each subsystem test set performs its own self-test and reports its status to the STC/CDCS. The STC/CDCS, in turn, performs an internal self-test and, for launch checkout operations, is programmed to test similarly the various adapters, the Planetary Vehicle monitor console, and the JSCR, and LCE support facility equipments.

3.2.5.2 LCE Functions

In this section the functional descriptions of the LCE are categorized and arranged as follows:

1) Up-link - A description of the kinds of controls and commands provided and the paths over which they flow;
2) Down-link - A description of the ways in which data are received from the spacecraft and processed, and the various display capabilities provided;
3) Power control;
4) Environmental control;
5) Emergency communications system;
6) Recording and time-tagging.

Figure 3.2-5 contains information pertinent to these functional descriptions of the LCE.

Up-Link--Information flowing through the up-link to the spacecraft is classified into two types, that is, discrete controls and radio commands. Controls are transmitted via hardline and cause specific events to occur in the spacecraft. Radio commands reach the spacecraft via either the link or a coax in the umbilical. More detailed information is given below regarding the controls and commands.

Discrete Controls--Controls are available at the Planetary Vehicle Monitor Console (PVMC) in the Blockhouse and at the STC, located in the SCF. The PVMC is connected into the STC Central Data and Control System and provides limited control of discrete events (examples are given later) for both Planetary Vehicles. Controls are either automatic or are implemented through the use of on-off type switches or pushbuttons. Signals are routed through the JSCR to the appropriate spacecraft subsystem.

The STC provides primary control for conditioning and testing the Planetary Vehicles. All controls available at the PVMC are also provided in the STC. Control at the STC is provided by the means described in Section 3.1.5.2.

Examples of spacecraft discrete control functions performed by the STC are as follows:

1) Power control signals to apply or remove power, and to switch over from ground power to spacecraft power;

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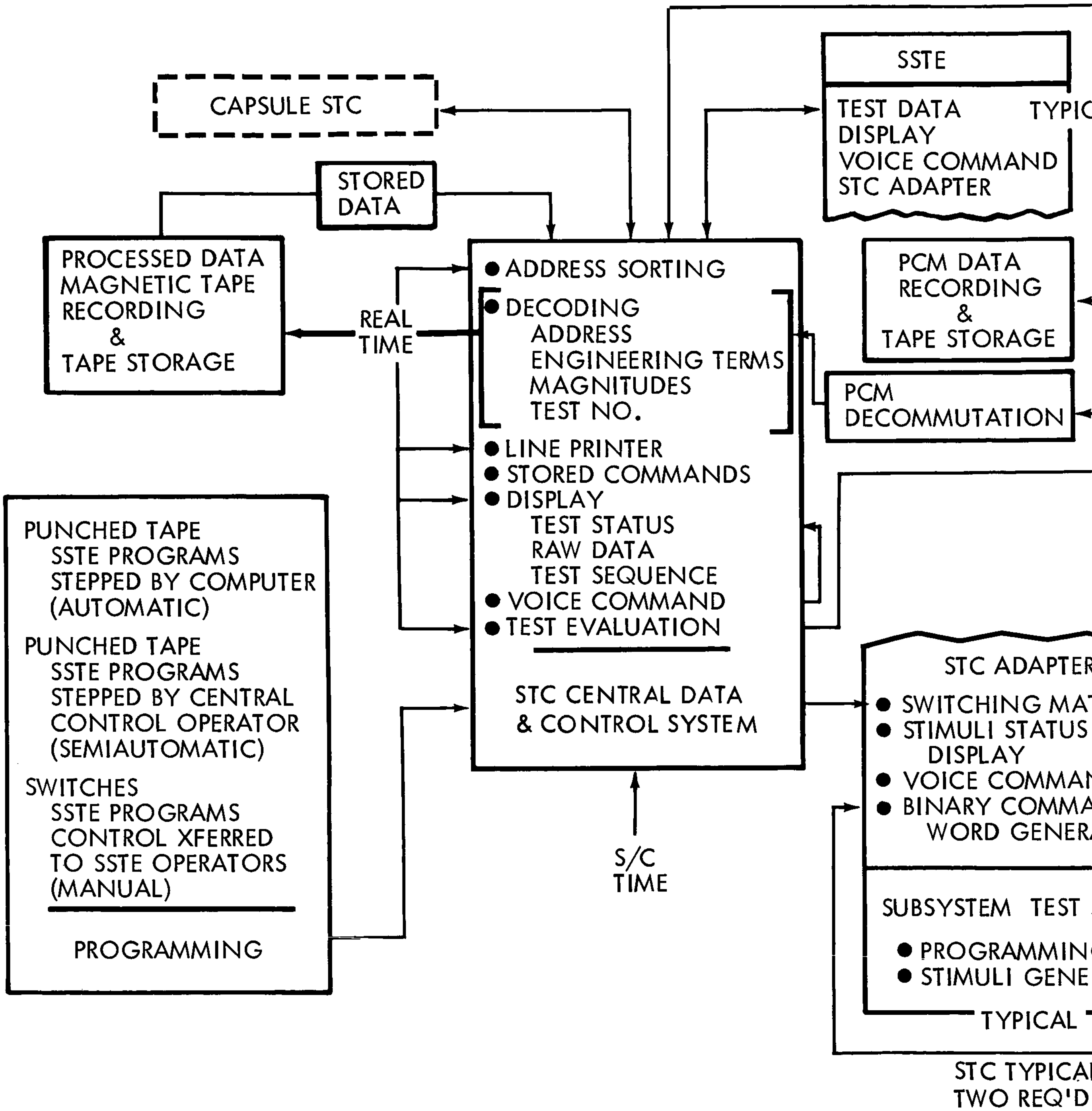
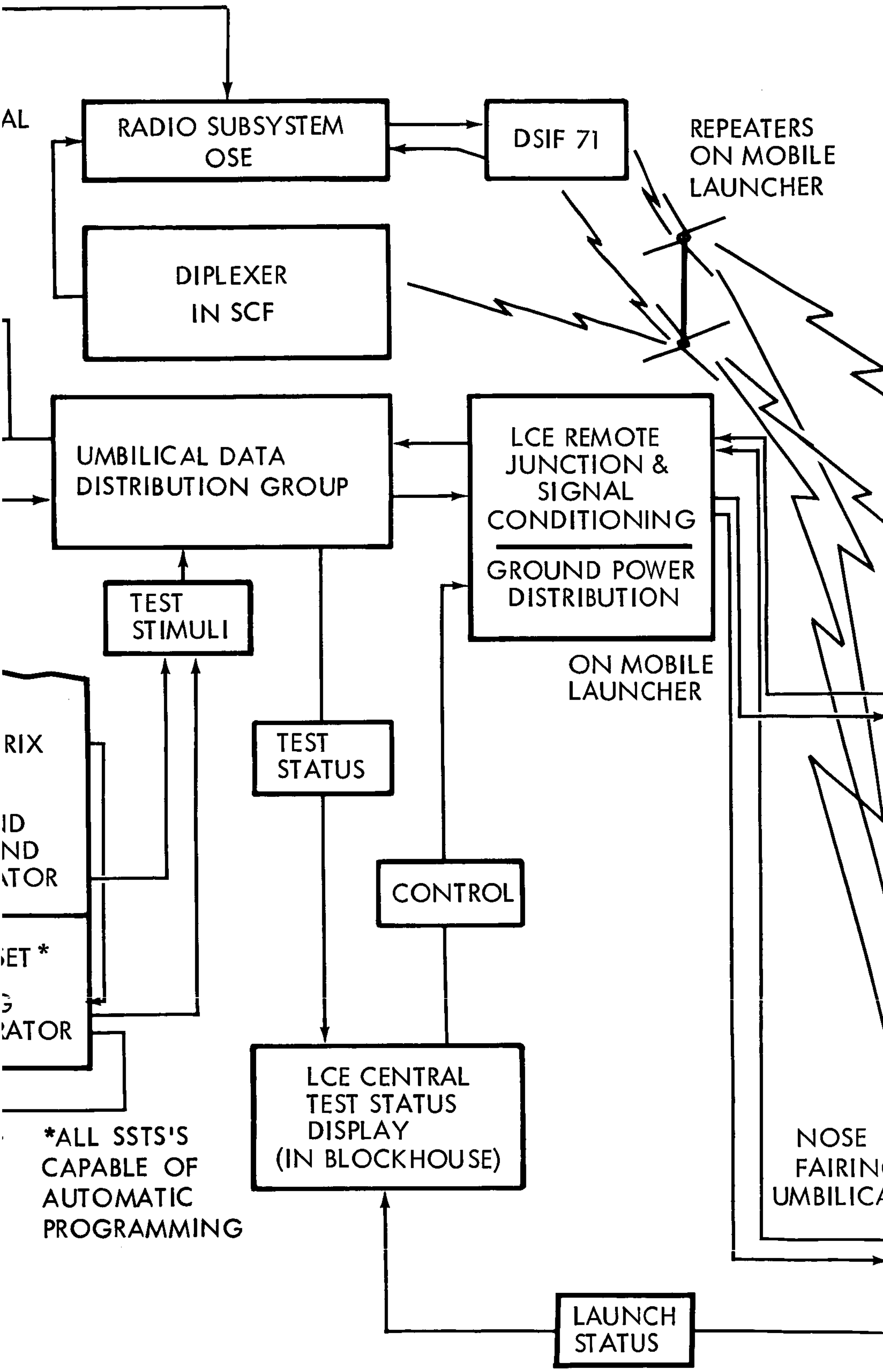


CAPSULE STC
STORED DATA
PROCESSED DATA MAGNETIC TAPE RECORDING & TAPE STORAGE
REAL TIME
SSTE
TEST DATA DISPLAY
VOICE COMMAND
STC ADAPTER
TYPIC
PCM DATA RECORDING & TAPE STORAGE
PCM DECOMMUTATION
ADDRESS SORTING
DECODING
ADDRESS
ENGINEERING TERMS
MAGNITUDES
TEST NO.
LINE PRINTER
STORED COMMANDS
DISPLAY
TEST STATUS
RAW DATA
TEST SEQUENCE
VOICE COMMAND
TEST EVALUATION
STC CENTRAL DATA & CONTROL SYSTEM
PUNCHED TAPE
SSTE PROGRAMS
STEPPED BY COMPUTER
(AUTOMATIC)
PUNCHED TAPE
SSTE PROGRAMS
STEPPED BY CENTRAL CONTROL OPERATOR
(SEMIAUTOMATIC)
SWITCHES
SSTE PROGRAMS
CONTROL XFERRED TO SSTE OPERATORS
(MANUAL)
PROGRAMMING
S/C TIME
STC ADAPTER
SWITCHING MA
STIMULI STATUS DISPLAY
VOICE COMMA
BINARY COMMA WORD GENER
SUBSYSTEM TEST
PROGRAMMIN
STIMULI GENE
TYPICAL
STC TYPICA
TWO REQ'D

0



AL
RADIO SUBSYSTEM OSE
DSIF 71
REPEATERS ON MOBILE LAUNCHER
DIPLEXER IN SCF
UMBILICAL DATA DISTRIBUTION GROUP
LCE REMOTE JUNCTION & SIGNAL CONDITIONING
GROUND POWER DISTRIBUTION
ON MOBILE LAUNCHER
TEST STIMULI
RIX
ID
ND
TOR
TEST STATUS
CONTROL
ET *
ATOR
LCE CENTRAL TEST STATUS DISPLAY (IN BLOCKHOUSE)
*ALL SSTS'S CAPABLE OF AUTOMATIC PROGRAMMING
NOSE
LAUNCH STATUS

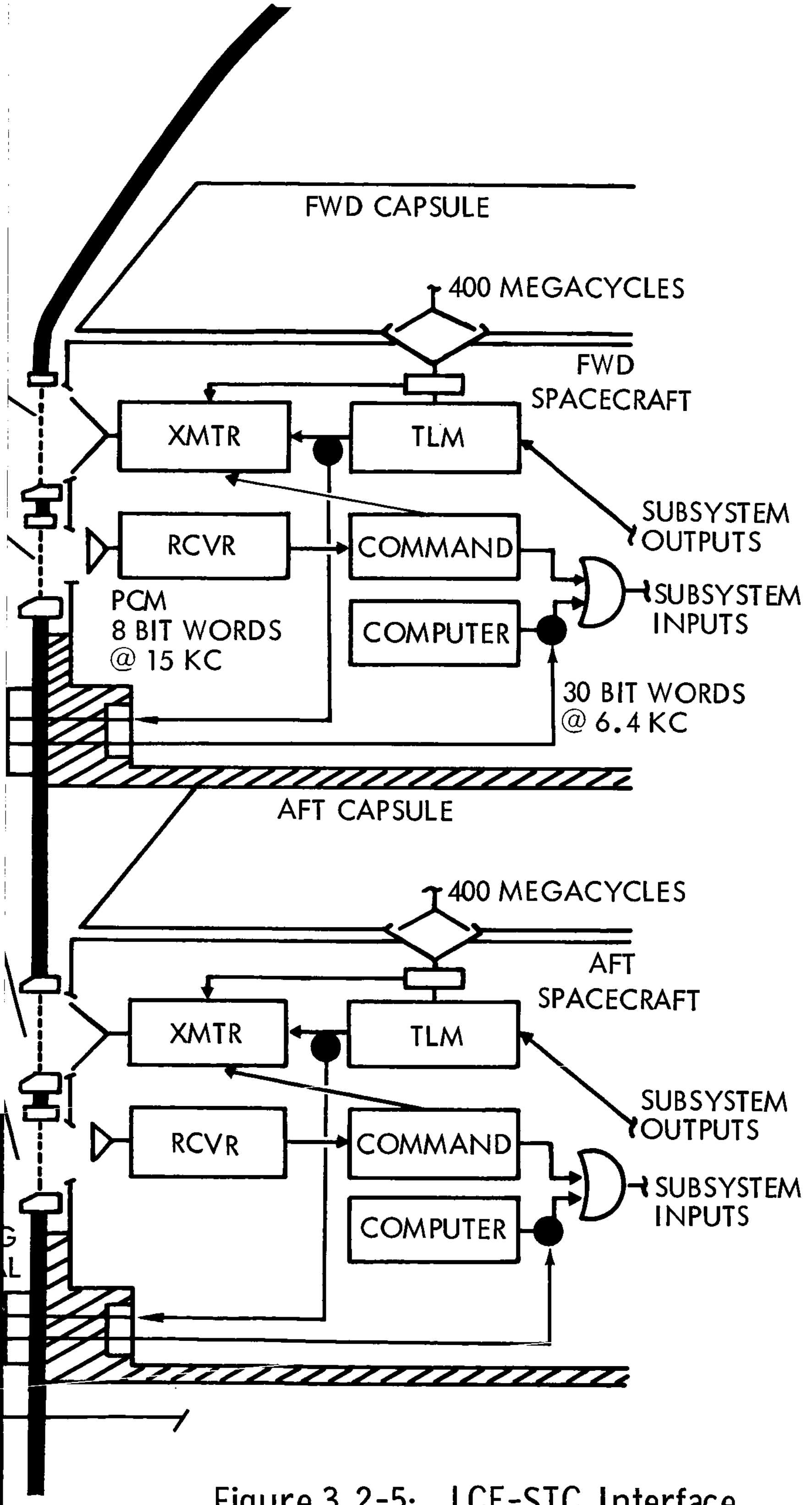


Figure 3.2-5: LCE-STC Interface

3

2) Automatic emergency control signals, derived from the emergency sensors, applied to the Spacecraft to preclude all potential catastrophic events;

3) Control of selected spacecraft functions for automatically conditioning the spacecraft to a safe mode in the event of power outages;

4) A manual control of the HOLD COUNT--RETURN TO SAFE CONDITION command, which provides a contingency control in the event that uncontrollable conditions disable the redundant, automatic safing circuitry.

Radio Commands--The radio command data are generated only in the STC CDCS. The various automatic commands, stimuli, and sequence initiations which are part of the countdown are automatically generated in the proper format within the STC under the control of countdown sequence computer subroutines. This control is synchronized with the central timing and countdown sequence for the entire space vehicle. The subroutines are capable of isolating any spacecraft malfunction to a particular subsystem. The countdown sequence program is constructed to permit manual initiation of various sub-sequences during simulated countdowns.

The commands thus generated in the CDCS are then routed to the radio subsystem OSE and the spacecraft checkout facility junction and interface equipment rack. The radio subsystem OSE modulates the up-link command stream, which is then presented to the diplexer for radiation by the antenna.

Commands are transmitted to the Planetary Vehicle via the SCF antenna route for the primary purpose of testing the Planetary Vehicle rf link. Use of the rf link for ground testing is kept to a minimum because of the low (1 bps) rate of accepting commands in the command detector.

The spacecraft checkout facility junction and interface equipment rack routes the command, via hardline, to the JSCR, which routes it to the spacecraft via the umbilical. This furnishes a by-pass of the command detector and permits command rates up to 4.8 kbps, and it provides a direct-access, nonradiating path to the Planetary Vehicle for commands generated in the STC.

The data, sync, and on-lock signals are carried by three coaxial lines into the input of the command decoder. By bypassing the command detector which limits the input rate to 1.0 bps, commands may be received at a 4.8 kbps rate, improving test flexibility and permitting accelerated loading of the C&S memory. One additional command input, via coax into the radio subsystem, is used to test that subsystem in the absence of an rf link.

Down-link--The description of the down-link information flow and the processing of that information is categorized and arranged as follows:

1) Telemetry Monitors--A description is given of the various routes over which telemetry information flows; the purpose for which each route is exercised is also included.
2) Hardline Monitors--The continuously monitored critical spacecraft functions are identified.

3) CDCS Data Processing--A description is given of the processing and the routing of the processed information.

4) Displays--The various types of displays are described.

Telemetry Monitors--Telemetry data from both Planetary Vehicles are transmitted both open-loop and closed-loop to the respective STC's. Open loop data are routed through an S-band double-ended repeater (SBDER) antenna mounted on the M/L to the STC via DSIF 71 and via the antenna mounted on the SCF. Closed-loop data are routed via the umbilical.

1) Open-Loop DSIF 71 Route

   The primary purposes of the rf link from the M/L to the SCF via DSIF 71 are to check out compatibility between the Planetary Vehicles and MOS and to verify operational readiness. The two rf signals received at DSIF 71 are processed there in a manner identical to the operational methods and are then routed to the central data and control system in the STC for further processing. During countdown this route is utilized to direct pertinent information to the mission operations center to keep the mission director apprised of Planetary Vehicle status.

2) Open-Loop SCF Route

   The primary purpose of the rf link from the M/L to spacecraft checkout facility is to verify operational readiness of the entire rf propagation system in the spacecraft. At the SCF a diplexer separates the two rf signals received at different frequencies from the Planetary Vehicles and routes them to the radio subsystem OSE.

The radio subsystem OSE demodulates the down-link telemetry stream received from the diplexer and routes the modulated subcarrier to the CDCS in the STC.

3) Closed Loop

The primary purpose of the closed-loop link is to obtain spacecraft telemetry data without requiring rf radiation. Modulated subcarrier telemetry signals from both Planetary Vehicles are routed through separate umbilical coax paths, through the M/L junction boxes, across separate high-speed-data-links, to their respective CDCS units, where they are processed as was the data which flowed across the open-loop routes.

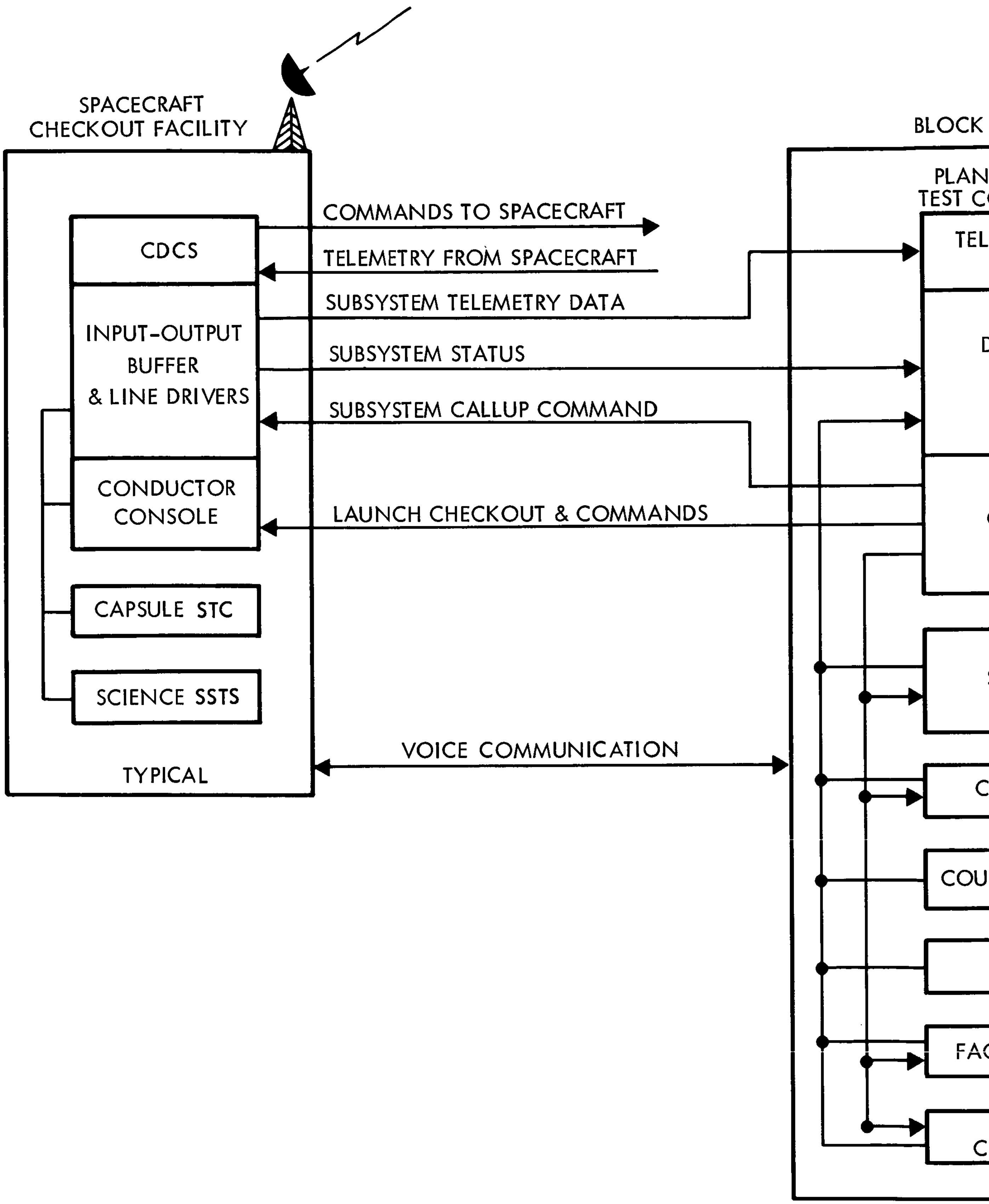
<u>Hardline Monitors</u>--Hardlines are provided to permit continuous monitoring of critical spacecraft functions. These signals, identified in Figure 3.2-6, are routed to the JSCR, where evaluation takes place. The resulting status information is sent to both the PVMC in the B/H and to the CDCS in the SCF. In a no-go condition, the signal routed to the CDCS (over redundant paths) causes appropriate corrective action to be taken. These critical function signals are evaluated in the JSCR to optimize the success path for detecting and acting on any critical malfunction.

In addition to the hardline monitors described above, the following support functions are monitored both at the PVMC and the STC:

1) ETR time

2) Countdown time

3) LCE facility power status



SPACECRAFT
CHECKOUT FACILITY
CDCS
INPUT-OUTPUT
BUFFER
& LINE DRIVERS
CONDUCTOR
CONSOLE
CAPSULE STC
SCIENCE SSTS
TYPICAL
COMMANDS TO SPACECRAFT
TELEMETRY FROM SPACECRAFT
SUBSYSTEM TELEMETRY DATA
SUBSYSTEM STATUS
SUBSYSTEM CALLUP COMMAND
LAUNCH CHECKOUT & COMMANDS
VOICE COMMUNICATION
BLOCK

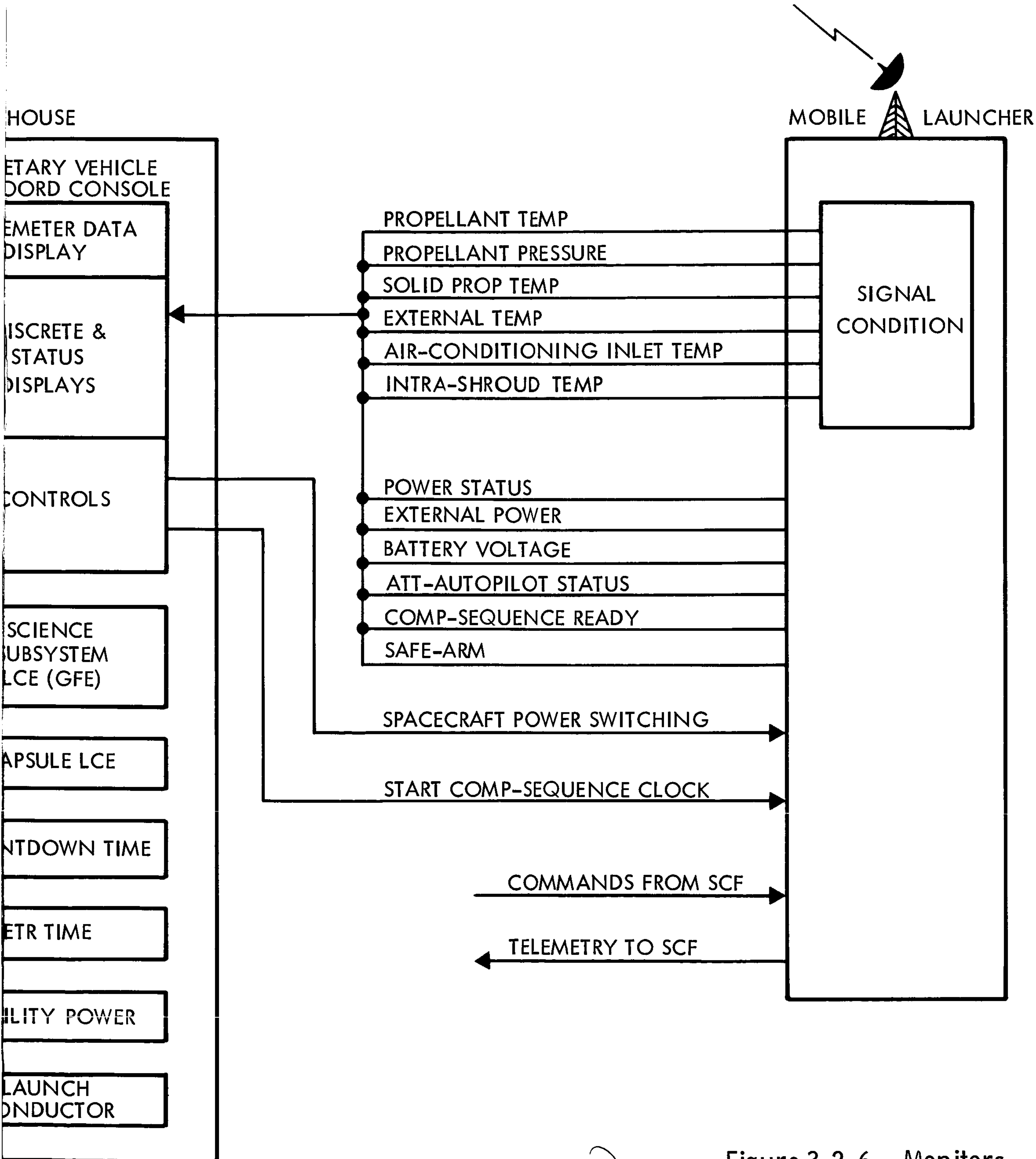


Figure 3.2-6: Monitors

CDCS Data Processing--In the CDCS the data received from either the closed-loop or the open-loop routes are decommutated, decoded, formatted, calibrated, evaluated and presented for display at the various subsystem test adapters. This information, containing the status and responses of the Planetary Vehicle subsystems to the commands and stimuli applied during the countdown sequence, is also time-tagged and recorded. Selected status information on the Planetary Vehicle subsystems is routed to the PVMC in the blockhouse to keep the Planetary Vehicle test coordinator apprised of Planetary Vehicle status. Software for LCE operations is discussed in Section 3.6.5.2.

Displays--Displays are provided in three areas; namely, the B/H, the explosive safe area (ESA), and the SCF. The SCF displays are described in Section 3.1.5.2. The blockhouse and ESA displays are on the PVMC and are described below.

The PVMC displays essentially four kinds of data. They are: discrete-status information, launch-peculiar alphanumerics, meters and strip-chart recordings, and auxiliary data. The discrete-status displays are bistatic devices, such as lights, which indicate the condition of selected launch countdown functions. The alphanumerics present such information as launch countdown time and ETR time. Meters and strip-chart recorders are provided for displaying analog signals, such as spacecraft temperatures and pressures, and event signals, such as umbilical disconnect. The auxiliary data are essentially a duplicate of the information presented to the specialists at the subsystem test adapters in the STC. This information is available to the Planetary Vehicle test coordinator on a call-up basis, and is subject to a priority interrupt.

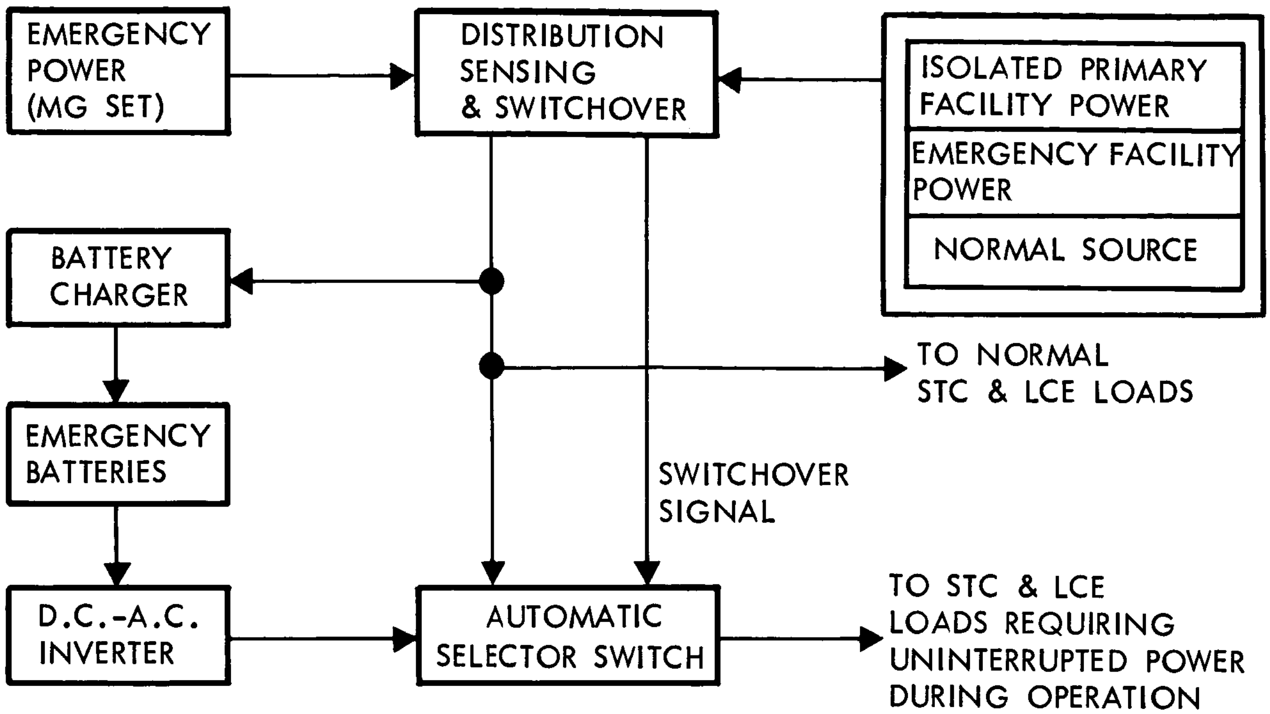
Power Control--Figure 3.2-7 shows the power supplies required at the various LCE locations to support the tests.

1) Normal main power--Refer to Sections 3.4.5.7 and 3.4.6.4.
2) Emergency main power--Refer to Sections 3.4.5.7 and 3.4.6.4.
3) Spacecraft external power--Spacecraft ground power is provided by the power subsystem test set which has an output capability of several AC and DC voltages as described under Section 4.2.1. These outputs to the spacecraft are controlled by the CDCS.
4) Emergency external power for spacecraft--In the event both main power sources fail, an emergency battery is automatically switched into the system to provide power for control and indications of selected spacecraft functions.

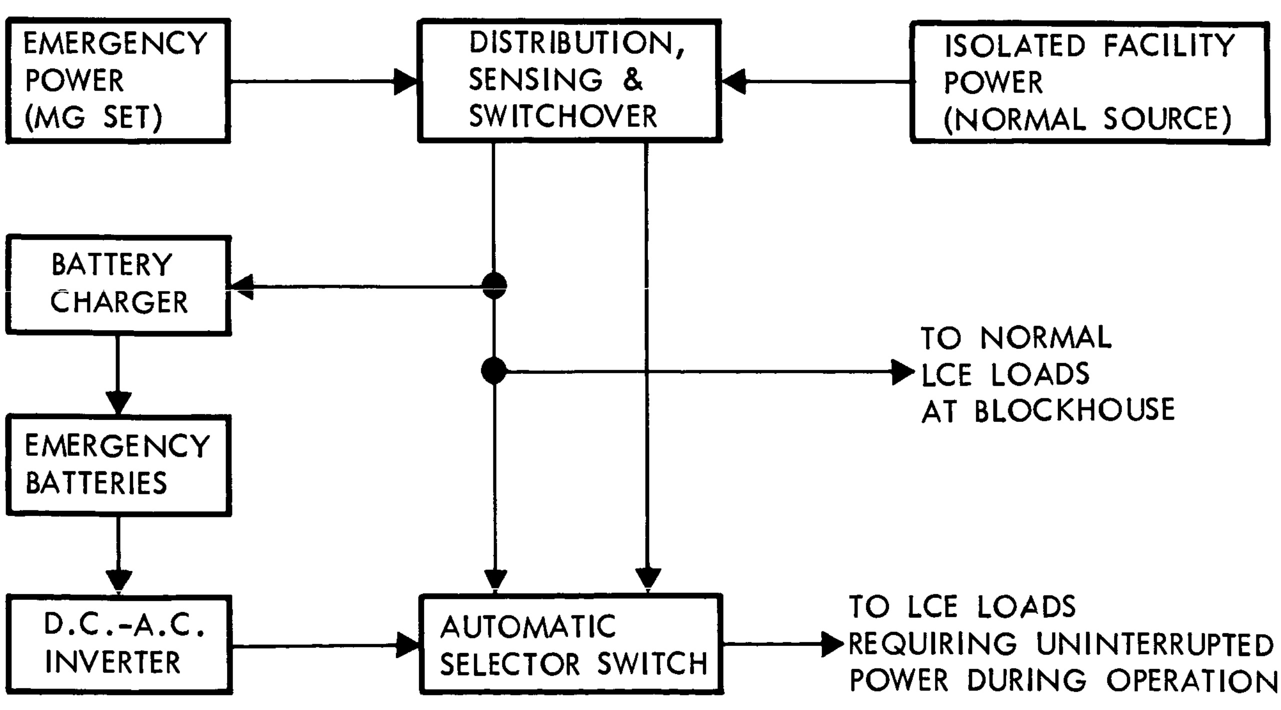
Environmental Control

1) Spacecraft Environmental Control

   A controlled environment for each Planetary Vehicle is provided by the Launch Vehicle System. However, the temperature of each S/C is monitored by the LCE and, if conditions warrant, a request is transmitted to the appropriate authority for corrective action.
2) LCE Environmental Control
   a) Units of LCE are designed to connect into the existing air conditioning and distribution system of the facility, i.e., the SCF, ESA and blockhouse. Units are designed to function at above normal temperatures for short periods of time in order to minimize probability of failure of a mission due to loss of air conditioning. Each unit is provided with a visual temperature level alarm.



POWER DISTRIBUTION EQUIPMENT AT THE SCF AND ESA



POWER DISTRIBUTION EQUIPMENT AT BLOCKHOUSE

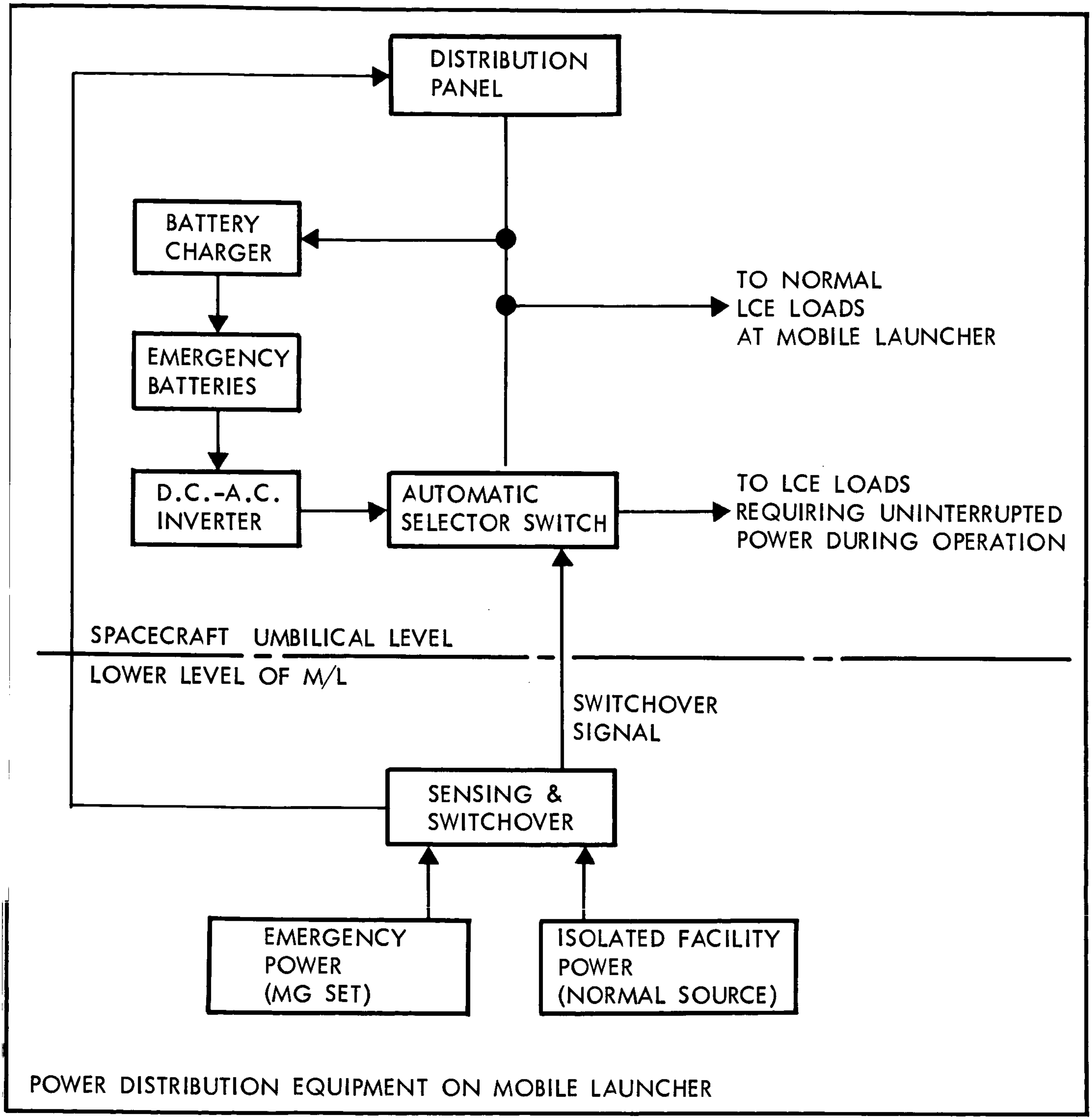


Figure 3.2-7: STC-LCE Power Distribution Block Diagram

b) Again refer to Section 3.1.5.2 for description of STC environmental control provisions.

Emergency Communications System--A voice communications link between the launch pad, the PVMC in the blockhouse and the planetary operations control center is provided for emergency communications. Special panels at the launch pad and at the blockhouse will be connected to the planetary operations control center by a telephone company lease line. Lifting the handset off the cradle at any facility sounds a strident alarm at the other two facilities. To emphasize the function, the panels are conspicuously colored.

Recording and Time-Tagging--A time-tagged recording is made of all data and power signals which enter or leave the LCE. In addition, appropriate processed data flowing between LCE components at the various locations are time-tagged and recorded.

To accomplish this, two recorder racks are located at the B/H (or ESA) in addition to the recording capability already provided in the STC.

3.2.5.3 Support Equipment and Facilities Requirements

Junction and Interface Equipment Racks--

M/L Junction and Signal Conditioning Rack (JSCR)--One JSCR is provided for each Planetary Vehicle. In addition to providing for signal distribution, the JSCR also houses the signal conditioners, power switching and control, evaluation circuitry for critical functions, and the emergency sensing equipment.

PTCR J-Box--Since the Pad Terminal Connection Room (PTCR) is an interface point for all signals going to or leaving equipment and vehicles located on the M/L, a junction box provides the flexibility required to mate with existing cabling runs.

Blockhouse Junction and Interface Equipment Rack--This rack provides the required flexibility for mating interfaces from the PTCR, the LV ESE, central timing, and SCF with LCE equipment located in the B/H. In addition, this rack performs similar functions at the ESA when the LCE is used to support testing at that location.

SCF Junction and Interface Equipment Rack--This rack provides the flexibility required to adapt the STC to the LCE at the various test locations.

Launch Vehicle ESE Simulator--A launch vehicle ESE Simulator is provided for use at the ESA. Certain functions which are ordinarily supplied by the launch vehicle ESE, such as countdown time, are provided by the simulator since no launch vehicle ESE exists at the ESA. Launch Vehicle ESE circuits which are nonfunctional during ESA testing are replaced by dummy loads.

In addition to testing at the ESA, the simulator is used during spacecraft-LCE compatibility testing when no launch vehicle ESE is available.

Voice Communications--Between all test consoles and all test locations.

Switching and Distribution Panel--This panel will normally provide facility main power but, on failure or out-of-tolerance condition, will automatically switch to the emergency main power.

Interlocation OSE Cabling--

1) Spacecraft Checkout Facility - Magnetic Mapping Facility;

2) Spacecraft Checkout Facility - Explosive Safe Area;

3) Spacecraft Checkout Facility - DSIF;

4) Spacecraft Checkout Facility - Blockhouse;

5) Blockhouse - Mobile Launcher.

3.2.6 Interfaces

Figure 3.2-8 contains information pertinent to these interface descriptions.

3.2.6.1 Pad Terminal Connection Room (PTCR)

The PTCR is an interface point for all signals going to or leaving equipment and vehicles located on the mobile launcher. No function other than routing is performed on the signals.

3.2.6.2 Capsule OSE

The STC extracts from the telemetry stream pertinent information for routing to the capsule STC and capsule LCE; this capsule OSE serves functions for the capsule which are very similar in nature to the functions served for the spacecraft by the spacecraft OSE. The signals across this interface are in digital format and constitute the total extent of information flowing to and from the capsule, since capsule down-link data are contained in the spacecraft telemetry stream and commands are routed to the capsule through the spacecraft CC&S subsystem or, as a back-up through the command subsystem.

3.2.6.3 Facilities

Facility power is provided on the mobile launcher to the JSCR. This power is monitored and recorded by the LCE located in the blockhouse.

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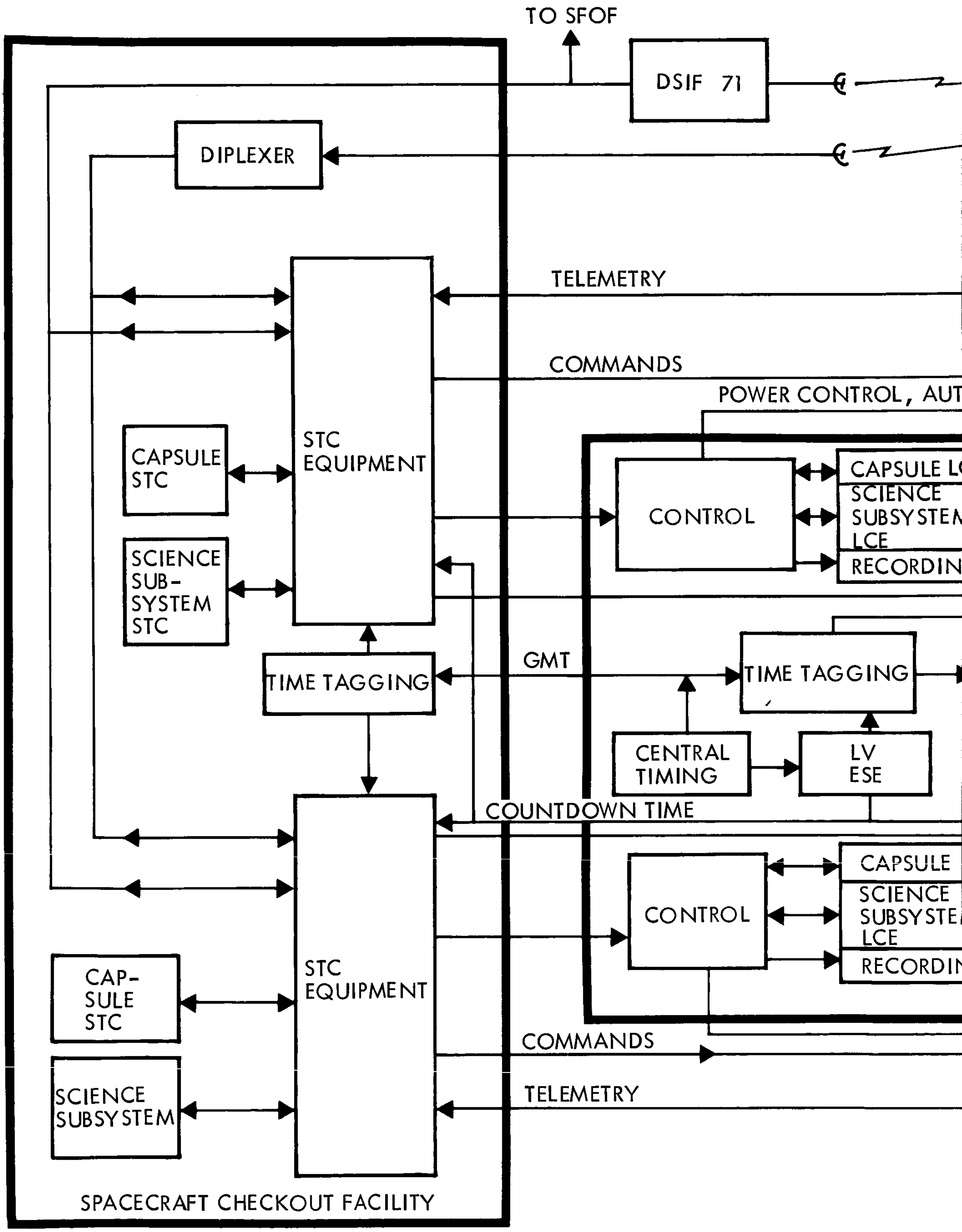
TO SFOF
DSIF 71
DIPLEXER
TELEMETRY
COMMANDS
POWER CONTROL, AUT
CAPSULE STC
STC EQUIPMENT
CONTROL
CAPSULE L
SCIENCE SUBSYSTEM LCE
RECORDIN
SCIENCE SUB-SYSTEM STC
TIME TAGGING
GMT
TIME TAGGING
CENTRAL TIMING
LV ESE
COUNTDOWN TIME
CONTROL
CAPSULE
SCIENCE SUBSYSTE LCE
RECORDIN
CAP-SULE STC
STC EQUIPMENT
SCIENCE SUBSYSTEM
COMMANDS
TELEMETRY
SPACECRAFT CHECKOUT FACILITY
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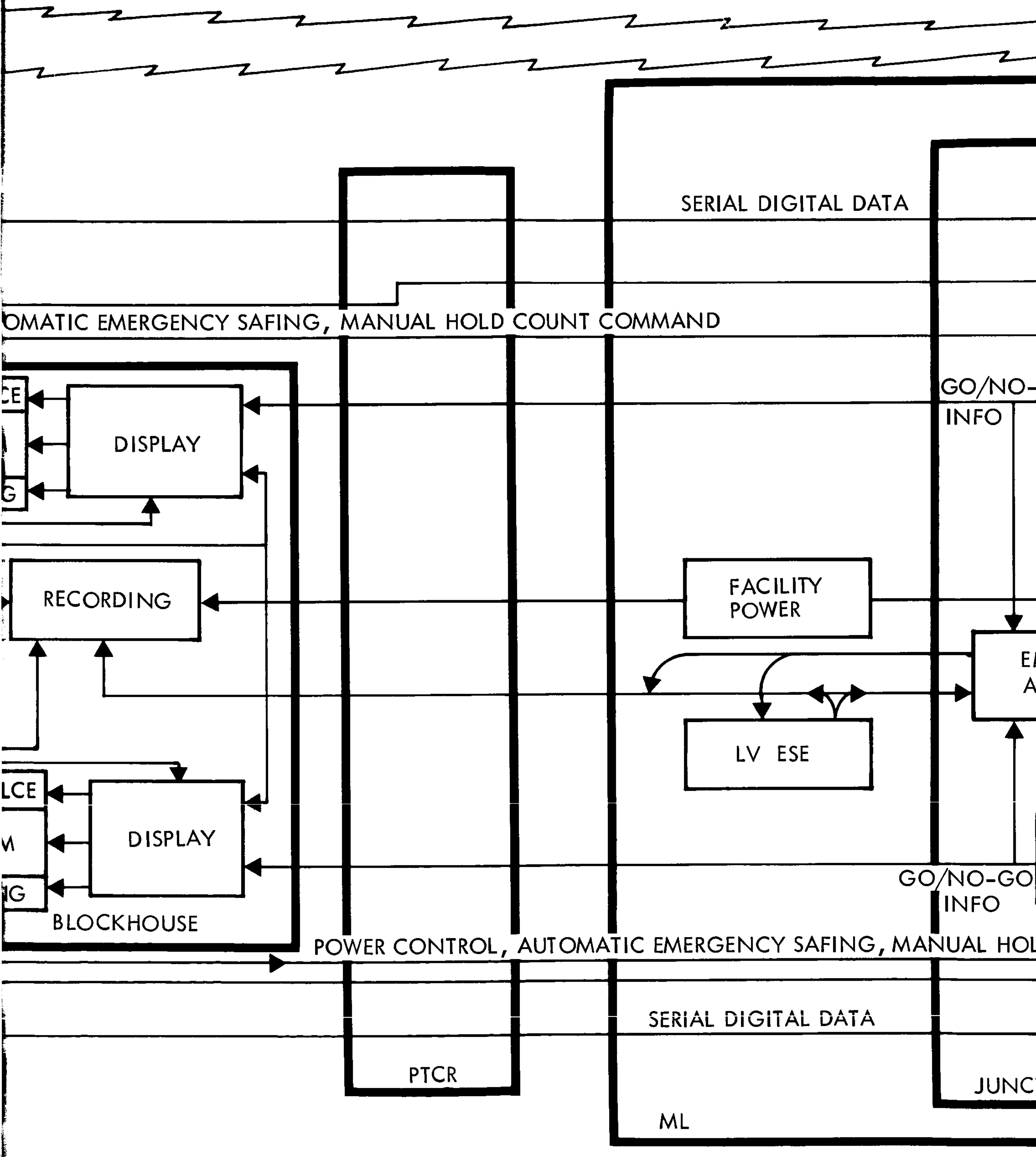

SERIAL DIGITAL DATA
OMATIC EMERGENCY SAFING, MANUAL HOLD COUNT COMMAND
GO/NO-
INFO
DISPLAY
RECORDING
FACILITY POWER
LV ESE
LCE
DISPLAY
GO/NO-GO
INFO
BLOCKHOUSE
POWER CONTROL, AUTOMATIC EMERGENCY SAFING, MANUAL HOL
SERIAL DIGITAL DATA
PTCR
JUNC
ML

2

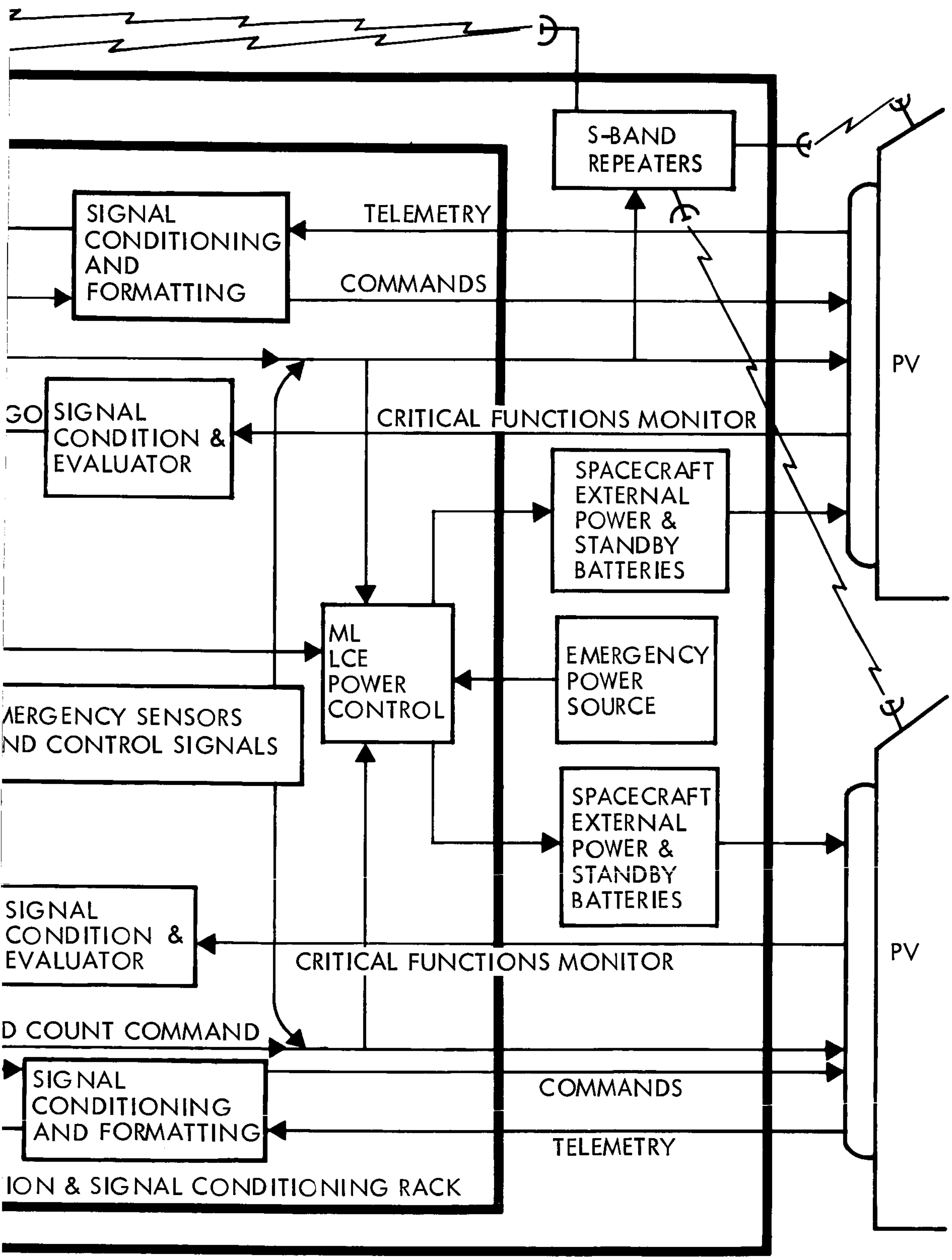


Figure 3.2-8: Launch Complex Block Diagram — LCE Usage

3.2.6.4 Central Timing/Countdown Sequence

Central timing signals are provided from Kennedy Space Center to the time-tagging equipment located in the blockhouse and in the SCF. In the absence of these signals, which are expressed in terms of GMT, the time code generator portion of the time-tagging equipment provides the timing signals.

The countdown sequence, generated by the launch vehicle ESE through the use of the central timing signals, is also provided to the time-tagging equipment and to the display equipment located in the blockhouse.

3.2.6.5 Launch Vehicle ESE

Emergency and Alarms--To provide a coordinated shutdown sequence in case of a no-go condition, the LCE must interface with the launch vehicle ESE. This interface is located on the mobile launcher. When a no-go condition is sensed by the monitors in the spacecraft, a signal is sent to the launch vehicle ESE to initiate a shutdown sequence. Similarly, when a no-go condition is sensed by the monitors in the launch vehicle, a signal from the launch vehicle ESE initiates the shutdown sequence of the spacecraft. Signal conditioning equipment required is minor because the interface signals are discrete.

Sequencing and Timing--Signals for sequencing and timing for the spacecraft during countdown are required from the launch vehicle ESE. This interface is located in the LCE/ESE interface rack at the blockhouse. The signals will be routed to the SCF and, therefore, require some amplification. The LCE is designed to accept the timing and sequencing signals provided, therefore, little conditioning equipment is required.

Status--Status of selected launch vehicle functions is required for display at the PVMC. Interface equipment required for converting the signals to an acceptable format is located in the LCE/ESE interface rack in the blockhouse. This equipment consists of signal conditioning circuits capable of converting the status signals to a compatible format.

Communications--A communications link is required between the spacecraft personnel and the launch vehicle personnel. Interface equipment required for this link consists of amplifiers, matching pads and distribution networks. STC personnel are tied into the communications link through the LCE.

Umbilical Interface Equipments--The LCE interfaces with the launch vehicle system at the ML end of the umbilical. The interface consists of a junction box which distributes umbilical lines to their appropriate destinations.

3.2.7 Performance Parameters

The performance parameters of the LCE are identified below in five categories: control, evaluation, monitors, display and recording. Capabilities of the STC are also used in conjunction with the LCE.

3.2.7.1 Control

The LCE has the capability to manually and automatically control the Spacecraft System and the capsule system both locally and from remote locations. LCE control is provided through direct access umbilical lines and through the command subsystems; each provides control of discrete events in the capsule system and the spacecraft subsystems. In addition, the LCE has the capability to control support functions as power switching, etc.

3.2.7.2 Evaluate

The LCE has the capability to receive and evaluate two PCM data streams at five different bit rates. Capability is also provided for simultaneously evaluating analog signals obtained through direct access umbilical lines from two planetary vehicles. Evaluation capability is provided for DSIF 71, LVS, and capsule system digital and analog interface signals.

3.2.7.3 Monitor

The LCE has the capability to monitor analog and discrete signals obtained through direct access umbilical lines for both Planetary Vehicles. These monitors are listed in Figure 3.2-6. In addition, the LCE has the capability to monitor interfacing systems for emergency conditions.

3.2.7.4 Display

The LCE has the capability to display Spacecraft System and capsule system discrete and analog signals both locally (near PV) and at remote locations. Capability is provided for displaying signals supplied from interfacing systems. Provision for monitoring the displays of the STC at remote locations is provided.

3.2.7.5 Recording (and time tagging)

The LCE has the capability to record and time-tag analog and digital umbilical signals for two Planetary Vehicles both locally and at remote locations. Capability for recording signals to and from systems interfacing with the LCE is provided. In addition, capability is provided for A-D conversion to enable recording of all analog signals in digital form.

3.2.8 Physical Description

The principal elements of the Launch Complex Equipment are located as shown earlier in Figure 3.2-2. Table 3.2-1 is a list of the equipment. The following comments are provided to supplement the information given in Figure 3.2-2 and Table 3.2-1.

The equipment contained within the SCF is described in Section 3.1.8.

The equipment mounted on the mobile launcher consists of the repeater antennas, the mobile launcher junction and signal conditioning rack, and the spacecraft external power supply. The JSCR and the power supply are suitably packaged to withstand the launch environment. The antennas are considered to be expendable under the environment of a launch.

The equipment located at the blockhouse is the same as that used at the ESA. This equipment consists of two racks of recording equipment, one junction and interface equipment rack, and one console of display and control equipment for the two planetary vehicles. The display and control characteristics have been described in Section 3.2.5.2.

The antennas mounted on the SCF provide continuity in the rf link between the mobile launcher and the STC. They are parabolic dishes with a line-of-sight alignment with the mobile launcher.
the mobile launcher.

3.2.9 Reliability and Safety

The LCE reliability is a mission reliability consideration. As noted in Section 3.17.3.16 of Volume A of this document, a goal of 0.99

Table 3.2-1: Launch Complex Equipment List

Set #1 consists of:

- Pad - Located Equipment

  Repeater Antenna
  M/L Junction and Signal Conditioning Rack
  Spacecraft External Power Supply
  Cabling
  Emergency Power
  PTCR Junction Box

- Blockhouse - Located Equipment

  Central Recording
  Planetary Vehicle Monitor Console
  B/H Junction and Interface Equipment Rack
  Cabling

- SCF - Located Equipment

  SCF Junction and Interface Equipment Rack
  Antenna/Diplexer
  Cabling
  STC Equipment

Set #2 is identical to set #1, except that:

- all equipment is located in the ESA
- an LV ESE Simulator is used as described in Section 3.2.5
- cabling to connect between the PTCR Junction Box and the B/H - Located equipment is required to replace that which is permanently installed at the pad.

reliability has been allocated to mission critical OSE. This reliability figure is defined as the probability that the OSE will successfully perform its required function throughout the mission, starting with the terminal countdown, and including thirty days of orbital operation. OSE success is defined as:

1) Performance without a failure causing spacecraft failure;
2) Performance without failure to detect designated spacecraft malfunctions when they occur;
3) Performance without a failure which would cause mission re-schedule after the start of terminal countdown.

Safety of both personnel and the spacecraft have been considered in the LCE design. General safety provisions are the same as those supplied to the STC and are listed in Section 3.1.9.

3.2.10 Design, Development, and Test Plan

This is the same as that described in Section 3.1.10.

3.3 ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT (AHSE), AND SERVICING EQUIPMENT

This equipment: (1) provides the capability of lifting, holding, and positioning Voyager vehicle assemblies and Operational Support Equipment (OSE) during assembly and test; (2) provides simulation and measurement to support these tests; (3) maintains environmental conditions including cleanliness and biological protection required by Voyager during transportation, handling, and storage; and (4) provides for transporting and handling Voyager assemblies and OSE.

It also includes items that supply gases and liquids to the spacecraft and provides for:

1) Decontamination of the internal hardware if the fluid systems associated with the orbit insertion motor and the reaction control system;
2) Sterilization of the freon and nitrogen to be loaded aboard the spacecraft.

3.3.1 Summary

AHSE and servicing equipment includes the following groups (described in Sections 3.3.5 through 3.3.17): measurement equipment--mechanical, test stands and fixtures, transportation equipment, safety equipment, work platform sets and access equipment, dollies, protective covers, shipping containers, lifting and handling devices, servicing equipment, installation kits and assembly equipment, environmental control equipment, and ancillary equipment.

AHSE hardware design was selected to be within the capability of present technology. Commercial equipment and components are used wherever possible. Techniques developed in the Ranger and Mariner programs are used on Voyager.

3.3.2 Applicable Documents

The following documents are used in defining AHSE and servicing equipment:

1) "Mariner Mars 1964 Spacecraft Environmental Test Results," June 15, 1965. (ED-268)
2) State and industrial safety codes, as applicable.

3.3.3 Design Requirements and Constraints

The derivation of AHSE design requirements and constraints through review of Voyager system specification documents, review of system engineering functional flow charts, and expansion of these flows, is discussed in following paragraphs.

3.3.3.1 Compliance with System Specifications

Except as noted in Section 3.3.3.2, JPL requirements and constraints are satisfied by the AHSE design.

3.3.3.2 Deviations

It is assumed that providing solar simulation meets the requirements of General Specification Paragraphs 4.3.2.15 and 4.3.3.1.12, which specify that a free-mode test shall be conducted using solar power on the propulsion test module (PTM) and all flight spacecraft. The present baseline provides that the free mode test will be conducted in the Kent space chamber using available solar simulation equipment. This preserves cleanliness, provides a space environment, and allows control of the test.

3.3.3.3 Derived Requirements

The requirements for the AHSE and servicing equipment are derived by expanding system engineering functional flow diagrams (an example of the functional flow diagrams is found in Section 2.4). Figure 3.3-1 is

Receive and Inspect Assembled Spacecraft Less Solar Panels Receive Solar Panels	→	Assemble Solar Panels to Spacecraft (1.8)	→	Flight Spacecraft Verification of Seattle Systems Reference Test

Activity Block 1.8: Assemble Solar Panels to Spacecraft

Activity Objective: Complete Flight Spacecraft assembly in preparation for systems reference tests

Subfunctions Performed:

Transport items from stores to assembly area.

Remove spacecraft from shipping container and place in spacecraft assembly stand.

Remove solar panels from shipping container.

Visually inspect all components.

Install solar panels.

Verify spacecraft configuration.

Align spacecraft as required.

Equipment Required:

- Spacecraft dolly
- Spacecraft shipping container
- Spacecraft sling
- Spacecraft aft handling adapter
- Spacecraft shipping container
- Spacecraft shipping cover
- Spacecraft shipping container shipping cover
- Spacecraft shipping container sling
- Precision vertical load adjusting device
- Spacecraft and planetary vehicle assembly stand
- Spacecraft assembly platform
- Spacecraft assembly kit
- Solar panel shipping trailer
- Solar panel handling frame
- Solar panel dolly
- Solar panel dust cover

Facility Requirements: Class-100,000 clean room atmosphere

FIGURE 3.3-1 ACTIVITY EXPANSION

a typical example of functional flow expansion. In this example, the function, "Assemble Solar Panels to Spacecraft," is first expanded to identify the subfunctions performed. The equipment and facilities required to accomplish these subfunctions are then identified.

Major equipment items identified by the above procedures are listed in Table 3.3-1.

3.3.4 Trade Studies

The requirements analysis and functional flow expansions described in Section 3.3.3.3 result in an AHSE baseline description.

The trade studies discussed in this section define equipment within the functional areas and confirm the end-item selection. The baseline end item is described and compared against possible alternates. Two viewpoint levels are discussed in evaluating the AHSE:

1) Major Trades--Where major spacecraft system concepts directly and significantly affect the OSE selection.
2) Design Trades--Where AHSE design level decisions are made in describing the baseline with a minimum effect on other spacecraft system elements.

These two trade categories are expanded in Sections 3.3.4.1 and 3.3.4.2.

3.3.4.1 Major Trades

Major AHSE trades completed in this study phase are:

1) Spacecraft transportation;
2) Free-mode testing;
3) Magnetic mapping.

Table 3.3-1: AHSE AND SERVICING EQUIPMENT REQUIREMENTS

	Usage Locations Seattle	KSC	Other
Measurement Equipment--Mechanical			
Weight and Balance Equipment	x	x	
Alignment Equipment	x	x	
Nose-Fairing Clearance-Measuring Equipment	x	x	
Orbit-Insertion-Motor Alignment Set	x	x	
Rate Table	x		
Test Stands and Fixtures			
Spacecraft/Planetary Vehicle Assembly Stand	x	x	
Inversion Stand	x		
Free-Mode Test Stand	x		
Spacecraft Positioner and Test Stand	x	x	
Encapsulated Planetary Vehicle Assembly Stand	x	x	
Propulsion Interaction Test Fixtures	x		
Planetary Vehicle Nose-Fairing-Separation Test Fixture	x		
Spacecraft Vibration-Test Fixture	x		
Science Subsystem Test Fixture	x	x	
Zero-g Test Equipment	x	x	
Capsule/Cannister Separation Test Fixture	x		
Subsystem TAT Fixtures and Equipment	x		
Subsystem Launch and Flight Environment TAT Fixtures	x		
Dynamic Loads Test Adapter	x		
Magnetic-Mapping Equipment	x	x	
Perm-Deperm Equipment	x	x	
Acoustic Test Fixture	x		
Propulsion Subsystem Combined Environment Test Equipment	x		
Propulsion Subsystem Drop, Burst, and Flow Test Equipment	x		
Planetary Vehicle Adapter Installation Stand	x	x	
Propulsion Subsystem Performance Verification Test Equipment	x		
Solar Panel Space Chamber Test Fixture	x		
Static Load Test Fixture	x		
Spacecraft Module Support Stands	x	x	
Transportation Equipment			
Spacecraft Transporter	x	x	
OSE Remote Site Transporter	x	x	
Solar-Panel Shipping Trailer	x	x	
Transportation Recording Equipment	x	x	

Table 3.3-1 (Continued)

	Usage Locations Seattle	KSC	Other
Safety Equipment			
Mechanical Protective Devices	x	x	
Personnel Protection Barriers	x	x	
Fueling Protective Devices	x	x	
Personnel Safety Equipment	x	x	
Safety Unit	x	x	
Work Platform Sets and Access Equipment			
Spacecraft Assembly Platform	x	x	
Spacecraft Positioner Work Platform	x	x	x
Goldstone Platform Set			x
Encapsulated Area Platform Set	x	x	
Capsule Installation Platform Set	x	x	
Simulator Usage Access Equipment	x	x	
Encapsulated Planetary-Vehicle/Launch-Vehicle Access Equipment		x	
OSE Operation Access Equipment	x	x	
Weight and Balance Area Platform Set	x	x	
Dollies			
Spacecraft Dolly	x	x	
General-Purpose Component Dollies	x	x	
Propulsion Module Dolly	x	x	
Solar Panel Dolly	x	x	
Protective Covers			
Spacecraft System Test Area Protective Cover	x	x	x
Spacecraft Protective Cover	x	x	x
Planetary Vehicle Protective Cover	x	x	
Encapsulated Planetary Vehicle Protective Cover	x	x	
Solar-Panel Dust Cover	x	x	
Spacecraft Shipping Cover	x	x	x
Spacecraft Shipping Container Shipping Cover	x	x	x
Shipping Containers			
Spacecraft Shipping Container	x	x	x
Planetary Vehicle Shipping Container	x	x	
Orbit-Insertion-Motor Shipping Container	x	x	
Louver Shipping Container	x	x	
Propulsion Module Shipping Container	x	x	x
Science Subsystem Shipping Containers	x	x	
Primary Reference and Autopilot Shipping Container	x	x	

Table 3.3-1 (Continued)

	Usage Locations Seattle	KSC	Other
Battery Shipping Container	x	x	
Planetary Sensor Shipping Containers	x	x	
Communications Subsystem Shipping Containers	x	x	
CC&S Shipping Container	x	x	
Pyrotechnic Subsystem Shipping Containers	x	x	
Electrical Power Subsystem Shipping Containers	x	x	
Data Handling Subsystem Shipping Containers	x	x	
Lifting and Handling Devices			
Spacecraft Aft Handling Adapter	x	x	
Spacecraft Forward Handling Adapter	x	x	
Spacecraft Sling	x	x	
Encapsulated Planetary Vehicle Lifting Equipment		x	
OSE Lifting Equipment	x	x	
Spacecraft Positioner Sling	x	x	
Orbit-Insertion-Motor Sling	x	x	
Chassis and Subchassis Handling Equipment	x	x	
Louver Handling Equipment	x	x	
Miscellaneous Component Lifting Equipment	x	x	
Precision Vertical Load-Adjusting Device	x	x	
Portable Installation Hoist	x	x	
Solar Panel Handling Frame	x	x	
Planetary Vehicle Adapter Lifting Set	x	x	
Decontamination Equipment Lifting Set		x	
Spacecraft Simulator Lifting Set	x	x	
Reaction-Control-Unit Handling Frame	x	x	
Orbit-Insertion-Motor Hoist	x	x	
Aircraft Loading/Unloading Equipment	x	x	
Spacecraft Shipping Container Sling	x	x	
Planetary Vehicle Shipping Container Sling	x	x	
Servicing Equipment			
Fuel Servicing Unit	x	x	
Propellant Decontamination Unit	x	x	
Nitrogen Servicing Unit	x	x	
Propulsion and Reaction-Control-Systems Test Set	x	x	
Thrust-Vector-Control Servicing Unit	x	x	
Decontamination Equipment		x	
Particle-Contamination Detector	x	x	
Installation Kits and Assembly Equipment			
Spacecraft Assembly Kit	x	x	
Planetary Vehicle Adapter-to-Fairing Installation Kit	x	x	

Table 3.3-1 (Continued)

	Usage Locations Seattle	KSC	Other
Capsule Installation/Demate Kit	x	x	
Propulsion Module Installation/Removal Kit	x	x	
Planetary Sensor Scan Platform Alignment/ Installation Kit	x	x	
High-Gain-Antenna Alignment/Installation Kit	x	x	
Component Installation Device	x	x	
Environmental Control Equipment			
Spacecraft Transportation Air Conditioning Unit	x	x	
Encapsulated Planetary Vehicle Environmental Control Unit	x	x	
System Test Cooling Unit	x	x	
Ancillary Equipment			
Dummy Solar Panels	x		
Dummy Capsule	x		
Orbit-Insertion-Motor Simulator	x		
Solar Simulator	x		
Auxiliary Solar Simulator	x		
Service Arm Simulator	x		
Solar Panel Illuminator	x		
Mechanical OSE Engineering Spacecraft Model	x		
OSE Magnetic Evaluation Equipment	x		
Subsystem Magnetic-Mapping Equipment	x		
Launch Environment Test Facility Equipment	x		
Pyrotechnic Shock Test Instrumentation Equipment	x		
Container Washdown Unit	x	x	
Antenna Pattern Test Equipment	x		
OSE Environmental Test Equipment	x		
Subsystem Module Support Stands	x		
Structural Test Model Fixture	x		
Reflectance Test Equipment	x		
Solar Panel Deployment Aids	x		

The results of these trades are summarized in Figures 3.3-2 through 3.3-5. In each case, the selected approach is reflected in the equipment description when an AHSE end item is affected.

Other system-level trades have been identified for completion in later study phases. The trades are listed in matrix form in Figure 3.3-6. These trades generally are selected to answer significant AHSE configuration and requirement questions after more detailed spacecraft and ground system designs have been completed. An AHSE baseline approach has been assumed in all cases.

3.3.4.2 AHSE Design Trades

Trade studies listed in this section are those performed at the AHSE design level. Figure 3.3-7 describes these trades and discusses the factors considered in selecting or confirming the AHSE baseline.

3.3.5 Measurement Equipment - Mechanical

This category includes items of AHSE required to verify weight, critical alignment, and critical clearances on the spacecraft. Major items include the weight and balance equipment set, the spacecraft alignment equipment set, the nose-fairing clearance measurement equipment set, and the orbit-insertion-motor-alignment set. Guidance and control subsystem measuring equipment (rate table) is described in Section 4.2.2.

3.3.5.1 Weight and Balance Equipment

The weight and balance equipment is used to determine Flight Spacecraft or Planetary Vehicle center of gravity (c.g.) in the X-Y plane, within

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: VOYAGER SPACECRAFT TRANSPORTATION--SEATTLE TO KSC MATRIX OF DESIGN APPROACH					SELECTION *				
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		Transportation Mode	Estimated Cost	Transit Time	Advantages	Disadvantages	Cost	Time	Degradation	Total	Rank
PURPOSE To determine the best method for transporting the Voyager spacecraft from Seattle to KSC.		I. ASSEMBLED SPACECRAFT									
		1. B-377PG-2 ("Super Guppy")	$65,000	2 days	Short transit time; NASA-contracted aircraft (A/C)	High Cost; Only one A/C available; Might require S/C shipment 90° to Z-axis	3	1	1	5	1
		2. Truck	76,000	20 days	Readily available	Long transit time; High cost; Travel restricted	3.5	5	2	10.5	8
		3. Ship	56,000	20 days	Moderate Cost; No size restriction	Long transit time; Cost based on volume; One scheduled trip per mo.; Increased protection required	2.5	5	2	9.5	7
		II. DISASSEMBLED SPACECRAFT									
		1. C-5A	31,000	2 days	Low cost; Short transit time; Environment controlled	A/C not yet available; Military A/C	1.4	1	3	5.4	2
		2. B-377PG-2	65,000	2 days	Same as I.1	Same as I.1	3	1	3	8	4
		3. B-377PG ("Pregnant Guppy")	44,000	2 days	Same as I.1	Only one A/C available	2	1	3	6	3
		4. Truck	35,000	14 days	Low cost; Readily available	Long transit time; travel restricted	1.6	3.5	4	9.1	6
		5. Ship	56,000	20 days	Same as I.3	Same as I.3	2.5	5	4	11.5	9
		6. Railroad	22,000	10 days	Lowest cost; Readily available	Fairly long transit time; Special handling required; Higher possiblity of damage	1	2.5	5	8.5	5

*Selection Factors

1. Cost factor based on relative costs, which include $1400/day for 10-man test team for each day of transit time.
2. Time factor based on relative transit time in days
3. Degradation factor to evaluate expected effect of the mode and transit time on the spacecraft reliability. Modes are ranked sequentially with "1" assigned to mode resulting in least degradation

SELECTED APPROACH

I.1.
Assembled in B-377PG-2

Figure 3.3-2: Trade Study Summary

TRADE STUDY SUMMARY SHEET / SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: FREE-MODE TESTING MATRIX OF DESIGN APPROACH	SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		
It is necessary to determine if a free-mode test is required on the Voyager spacecraft and approach. A free-mode test has the following objectives: 1) To demonstrate the correct operation of the spacecraft on solar power. 2) To verify the functional integrity of the spacecraft in the absence of any support-equipment electrical connections. 3) To establish initial compatibility among the solar panels, their cabling, and the spacecraft. 4) To verify that the presence of the system test complex and associated cabling in previous tests has not affected spacecraft data readouts. The acceptance criteria for the free-mode test are: 1) The functional integrity of the spacecraft while operating on solar power and without any electrical connection to support-equipment will be established. 2) Sufficient telemetry data will be obtained to verify the correct operation of the power subsystem in the system environment using solar power. A free-mode test requires: 1) Spacecraft with flight solar panels 2) Solar stimulation source 3) STC/LCE 4) Isolation mount 5) Ground cable umbilical disconnect 6) Facilities and cleanliness capabilities	DISCUSSION 1. A free-mode test is specified in the JPL document, "Preliminary Voyager 1971 Mission Specification--May 1, 1965". 2. A free-mode test is specified in the JPL document, "Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification for-Preliminary, September 17, 1965". 3. JPL conducted free-mode tests on Mariner C and D in sunlight at Pasadena. The following approaches have been considered: 1. Conduct test in Seattle sunlight. 2. Conduct test at Table Mountain, Washington, sunlight. 3. Conduct test at Table Mountain, California, sunlight. 4. Conduct test in Boeing space chamber at Kent, Washington, using simulated sunlight. 5. Conduct test in Boeing space chamber lid room using simulated sunlight. 6. Conduct test in a new Boeing facility at Kent using simulated sunlight.	

Figure 3.3-3: Trade Study Summary

TRADE STUDY SUMMARY SHEET | SOURCE OF REQUIREMENT | TRADE STUDY NUMBER & TITLE: FREE-MODE TESTING MATRIX OF DESIGN APPROACH | SELECTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

Functional and Technical Requirements

Approach	Demonstrate operation on solar power	Verify Functional integraty without support equipment	Establish initial capability of solar panels	Verify That support equip. does not affect spacecraft	Clean room requirements	Schedule	Cost	Comments
1. Seattle Sunlight	Yes	Yes	Yes	Yes	Yes		$ 77,800	Few cloud-free days; New facility required
2. Table Mtn., Wash., Sunlight	Yes	Yes	Yes	Yes	No		1,341,000	New roads, fac, & transp equip required; No. of cloud-free days unpredictable
3. Table Mtn., Calif., Sunlight	Yes	Yes	Yes	Yes	No		1,550,000	Roads, fac, & transp req'd; No. of cloud-free days unpredictable
4. Space Chamber Simul Sunlight	Yes	Yes	Yes	Yes	Yes		Low	Min. flow time; Cannot deploy solar panels; Growth spacecraft not covered
5. Lid Room using Simul Sunlight	Yes	Yes	Yes	Yes	Yes		150,000 above 4	Modify lid room to 100,000-class clean room, perform structural modifications
6. New Fac using Simul Sunlight	Yes	Yes	Yes	Yes	Yes		738,800 over 4	

1 Alternate 5.e. is utilizing the lid room with the 4.e. solar simulator lid as a light source. This can be accomplished with 9 months lead time. Should be adapted if (a) growth versions of S/C prevented fitting in the chamber; or (b) if chamber scheduling becomes critical.

SELECTION

ILLUMINATION ADEQUACY
4 5 6 3 2 1

CLOUD FREE DAYS
3 2 1

MAJOR OSE
4 5 6 1 2 3

TRANSPORTATION
4 5 6 1 2 3

FACILITIES
4 5 6 1 2 3

FLOW TIME
4 5 6 1 2 3

DATA QUALITY
4 5 3 6 2 1

DELTA COST
4 1 5 6 2 3

SELECTED APPROACH

4.3
Trade #4
2-5695-0-148
1st Alternate
1 5.e
Test is required.

Figure 3.3-4: Trade Study Summary

<table>
<tr><td>TRADE STUDY
SUMMARY SHEET</td><td>SOURCE OF REQUIREMENT
Performance & Design Requirements
General Specification</td><td colspan="2">TRADE STUDY NUMBER & TITLE
DETERMINE LOCATION & METHOD OF MAGNETIC MAPPING FACILITY & OSE (SYSTEM LEVEL)
MATRIX OF DESIGN APPROACH</td><td rowspan="2">SELECTION</td></tr>
<tr><td colspan="2">FUNCTIONAL AND TECHNICAL
DESIGN REQUIREMENTS</td><td>EARTH AMBIENT MAPPING AT SEATTLE & KSC
1.</td><td>EARTH AMBIENT MAPPING AT KSC ONLY
2.</td></tr>
<tr><td colspan="2">Functional Requirements:

A means of perming and deperming the spacecraft and mapping its latent magnetic field must be provided. To obtain accurate space data, this must be accomplished as close to actual launch as feasible.

Design Requirements:

The Flight Spacecraft must be permed in a 25-gauss field and depermed by an alternating decreasing field from 50 gauss maximum to zero. The magnetometer must sense no more than one gamma residual magnetism from the flight spacecraft after deperming.

Method of Performing the Mapping:

The method of performing this magnetic mapping is as specified in Memo 2-5963-7, dated November 10, 1965, W. C. Galloway to R. K. Mills. Consequently, this subject is not a part of this trade.

Competing Characteristics:

The parameters to be considered in order of importance are:

1. Probability of success
2. Performance (no differences for this trade)
3. Cost
4. Contributions to subsequent missions (no bearing on this trade)
5. Additional 1971 mission capability (no bearing on this trade)</td><td>Consists of two areas, both alike, consisting of a mapping facility and a perming-deperming facility.

Discussion

Pro

1. Potential problems would have a higher probability of detection.
2. Able to sell the spacecraft* in Seattle. In other words, be able to prove a residual magnetic field no greater than one gamma.
3. Test team learning enhanced with a greater probability of completing the KSC test on schedule.
4. Alterations to spacecraft (if required could be performed at the place of manufacture).

Con

1. Highest Cost-requires two magnetic mapping facilities and two perming-deperming facilities.
2. Increases processing flow time.

* Possibly minus the propulsion subsystem.</td><td>Consists of one area only, consisting of a mapping facility and a perming-deperming facility.

Discussion

Pro

1. Requires only one magnetic mapping facility and perming-deperming facility.
2. Requires less flow time, if no latent faults develop.
3. Least cost.

Con

1. Would require spacecraft shipment back to Seattle if latent fields over one gamma are detected at KSC.
2. Would not be able to sell spacecraft in Seattle unless a waiver to the specification as a requirement for sell-off is obtained.</td><td>Probability of Succes
1, 2

Cost
2, 1

SELECTED APPROACH

(1)</td></tr>
</table>

Figure 3.3-5: Trade Study Summary

MAJOR TRADE STUDIES IDENTIFIED FOR FUTURE WORK		
EQUIPMENT AREA	ALTERNATE 1	ALTERNATE 2
PERM-DEPERM EQUIPMENT	SELECT COIL DIAMETER TO SEPARATION DISTANCE RATIO $> \frac{D}{2}$ PRO: REDUCED COST CON: LESS UNIFORM FIELD	DEPERM SPACECRAFT OR PV WITHOUT RANDOM VEHICLE ROTATION; USE EARTH FIELD CAN-CELING COILS AND EQUIPMENT ORIENTATION ALONG EARTH-FIELD VECTOR. PRO: 1) LESS CHANCE OF VEHICLE DAMAGE 2) GREATER DEPERM CONFIDENCE CON: GREATER COST
SPACECRAFT TRANSPORT	SHIP THE SPACECRAFT AND CAPSULE AS SEPARATE UNITS FROM SEATTLE TO KSC. PRO: 1) LESS CHANCE OF SPACECRAFT OR CAPSULE DAMAGE 2) FEWER AHSE END ITEMS	SHIP THE SPACECRAFT DISASSEMBLED. VARIOUS DISASSEMBLY LEVELS CAN BE CONSIDERED. PRO: 1) ALLOWS GREATER TRANSPORT FLEXIBILITY 2) REDUCES AHSE END ITEM COSTS 3) PERMITS USE AT MORE TEST AND OPERATION LOCATIONS 4) DOES NOT ADD TO COST BECAUSE DISASSEMBLED SHIPMENT MUST BE PROVIDED AS PART OF THE TEST AND SPARES PROGRAM CON: 1) TEST CONFIDENCE MUST BE RE-ESTABLISHED
ENCAPSULATED PV HANDLING, ASSEMBLY, TESTING AND TRANSPORT	ENCAPSULATE BOTH PV'S WITHIN ONE NOSE FAIRING. SIMPLIFIED VEHICLE DESIGN MORE COMPLEX OSE FACILITIES AND GROUND OPERATIONS	PARTIAL ENCAPSULATION AT ESA AND ASSEMBLY AT PAD
P/V = PLANETARY VEHICLE S/C = SPACECRAFT		

Figure 3.3-6: System Level Trades Affecting AHSE Selection

FUNCTIONAL AREA	BASELINE DESCRIPTION
Weight & Balance Equipment	Platform Scales (3 reqd) provided with suppo X-Y plane weight & c.g.
Inplant Mobility	Metal wheels on steel tracks
Spacecraft Support Structure for Test Stands, Transportation & Lifting	One common "handling adapter-aft" is used f support in handling transport & test stand
Spacecraft & Planetary Vehicle Shipping Container	One shipping container is used to transport spacecraft or the PV** between test locatio Seattle. An adapter is used to enlarge the accommodate the PV. The same container is port the S/C from Seattle to AFETR.
Spacecraft Positioner & Test Stand/Goldstone Test Stand	A semifixed test stand is provided to serve tions. The stand incorporates motion capab the S/C "Z" axis from horizontal to vertica
Inplant Vehicle or Test Stand Drives	Either air motor or electric motor drives d ment item usage or application.
Propellant Measuring Equipment	Platform scales are used to weigh propellan the S/C
Airlock Large Item Handling Dolly	The spacecraft dolly is used for movement o hardware items through assembly area or ESA applies for the orbit-insertion engine, nos capsule, spacecraft bus, and solar panels. excludes handling of the encapsulated PV.
Orbit Insertion Engine Handling	The orbit insertion engine is handled in th stands, the spacecraft dolly, and slings. is provided.
Spacecraft Mechanism Support During Alignment	Support cables from overhead frames attach in a manner similar to those of the deploym counterweights, or fixed-position links are support elements.
Spacecraft Handling Fittings & Attachments	The AHSE items attach to the S/C structure capsule or the adapter attach points.
Magnetic Mapping Test Stand	A dolly provides support and orientation ca the PV or the flight spacecraft. This doll lar track with dimensions set by the radius Boom. The track/dolly system maintains the tion relative to a test magnetometer locate center.

* S/C = Spacecraft
** PV = Planetary Vehicle

	ALTERNATE 1	
›rt stand to measure	Load cells in place of platform scales.	Additi scales "Z" ax
	Rubber tire wheels (solid or pneumatic)	Overhe
ɔr primary S/C* ‹HSE	Separate Components for each major AHSE item	
. either the flight ns at AFETR or container to also used to trans-	Separate units for shipping the S/C from Seattle and for handling the assembled S/C or PV at test locations.	Separa tions S/C mo
both stated func- ility of roll about l.	Separate stands for STC and Goldstone usage.	Portab facili
epending upon equip-	Select either air or electric to allow standard approach	Other cable
ts loaded aboard	Metering at the OSE unit (Flowmeter or calibrated tanks)	Load c
f all large flying airlocks. This e fairing components, This approach	Provide means of airlock passage so that vehicle change does not take place.	
e ESA with cranes, No special dolly	Purchase a dolly from the engine manu- facturer.	Design usage.
to the mechanisms ent aids. Springs, used in the	Provide floor stands that ajust to the required zero "G" mech- anism positions.	Provid limiti
using only the	Provide special ground handling points on the S/C structure	Provid capsul handli entire
pability for either y mates to a circu- of the magnetometer spacecraft's posi- d at the track	Provide a four-position straight ele- ment track system to allow positioning the spacecraft at points about the test magnetometer. A simplified support mechanism is provided on the dolly to allow roll about the magnetometers Boom axis.	

ALTERNATE 2	NOTES
onal fixture for use with or load cells to also measure is c.g.	
ad crane	
te units for the three func- of S/C shipping, assembled vement and P/V movement.	Combinations may be worked.
le stands not requiring ty installation.	
mechanical drives such as a or hydraulic drives.	
ells for weighing.	S/C point sensors in tanks may be used as a design alternate.
a special dolly for ESA	
e floor stands with force ng support devices.	Other schemes are possible, including vertical position- ing of the mechanism during alignment.
e handling points on the e cannister to allow ng and shipping of the P.V.	Transportation scheme of shipment from Seattle may require Alternate 2.

Figure 3.3-7: AHSE Design Trade Studies (3)

±0.1 inch, referenced to the Z-axis. The holding fixture portion of the equipment interfaces with the spacecraft at the handling adapter separation joint or at the Planetary Vehicle adapter hard points. The holding fixture is tilted to permit determination of the c.g. location along the Z-axis.

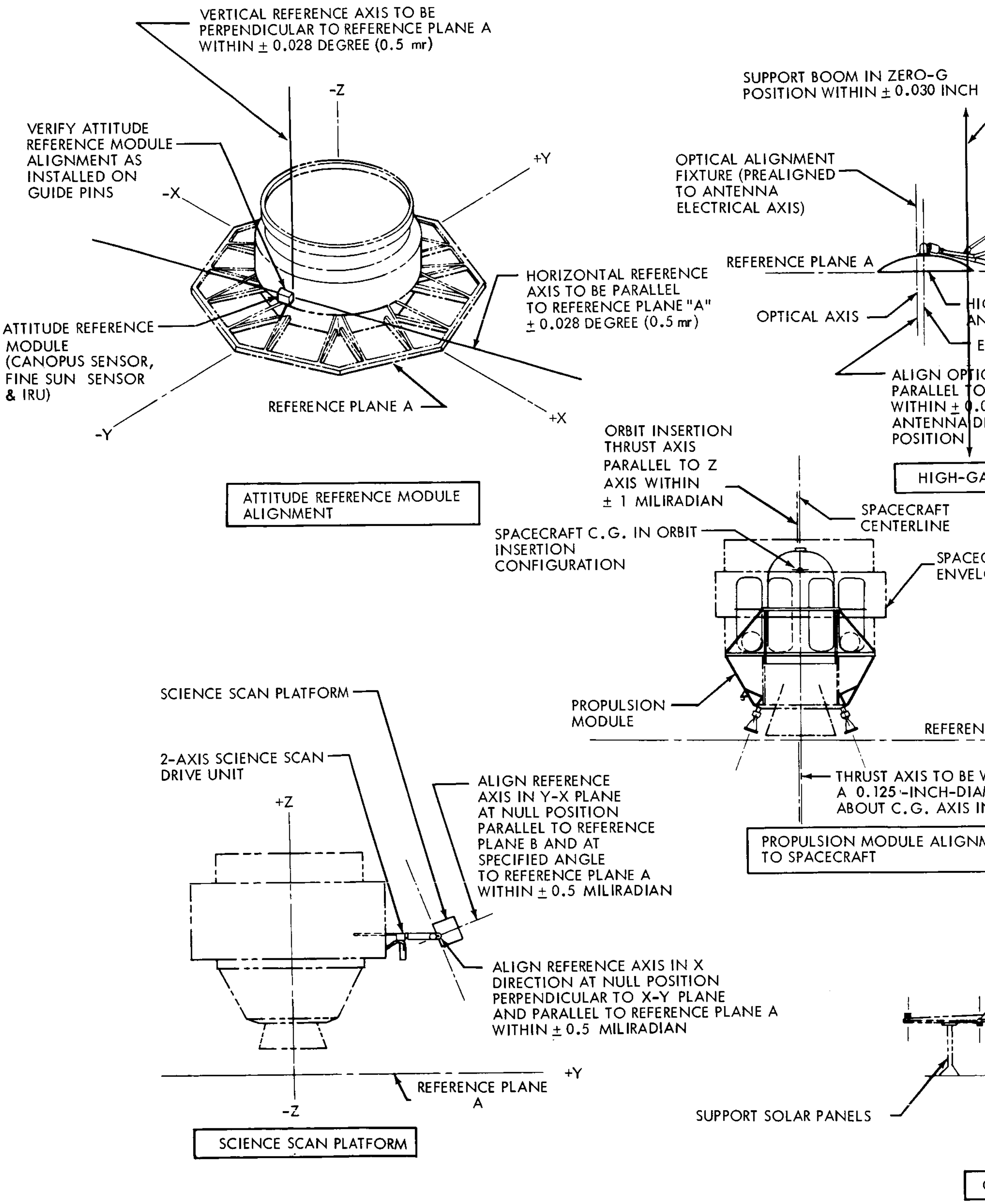
The holding fixture positions the spacecraft Z-axis vertical within ±2 minutes of arc. Reactions are measured within ±0.05 percent. The distance between weight points is known within ±0.010 inch.

Three platform scales or load cells are provided that rest on rigid supports at 120-degree intervals. The holding fixture has three pads with ball-socket mounting points. The fixture and spacecraft are supported by self-aligning pads and tooling balls on the scales. Leveling screws, and portable optical instruments are used to align the Z-axis of the spacecraft.

3.3.5.2 Alignment Equipment

The alignment equipment provides the capability to align all critical components of the spacecraft, including antennas, booms, scan platforms, solar panels, magnetometer, rocket motors, and Sun sensors. Figure 3.3-8 shows the Voyager system alignment requirements. Alignment is achieved by aligning the spacecraft to a planizer bed for a reference and then aligning components to the reference instruments on the planizer bed (see Figure 3.3-9).

Booms, antennas, and solar panels are supported in the position they will assume while deployed in a zero-gravity environment.


VERTICAL REFERENCE AXIS TO BE PERPENDICULAR TO REFERENCE PLANE A WITHIN ± 0.028 DEGREE (0.5 mr)
-Z
VERIFY ATTITUDE REFERENCE MODULE ALIGNMENT AS INSTALLED ON GUIDE PINS
-X
+Y
HORIZONTAL REFERENCE AXIS TO BE PARALLEL TO REFERENCE PLANE "A" ± 0.028 DEGREE (0.5 mr)
ATTITUDE REFERENCE MODULE (CANOPUS SENSOR, FINE SUN SENSOR & IRU)
REFERENCE PLANE A
-Y
+X
ATTITUDE REFERENCE MODULE ALIGNMENT
SUPPORT BOOM IN ZERO-G POSITION WITHIN ± 0.030 INCH
OPTICAL ALIGNMENT FIXTURE (PREALIGNED TO ANTENNA ELECTRICAL AXIS)
REFERENCE PLANE A
OPTICAL AXIS
ORBIT INSERTION THRUST AXIS PARALLEL TO Z AXIS WITHIN ± 1 MILIRADIAN
SPACECRAFT CENTERLINE
SPACECRAFT C.G. IN ORBIT INSERTION CONFIGURATION
PROPULSION MODULE
SCIENCE SCAN PLATFORM
2-AXIS SCIENCE SCAN DRIVE UNIT
+Z
ALIGN REFERENCE AXIS IN Y-X PLANE AT NULL POSITION PARALLEL TO REFERENCE PLANE B AND AT SPECIFIED ANGLE TO REFERENCE PLANE A WITHIN ± 0.5 MILIRADIAN
ALIGN REFERENCE AXIS IN X DIRECTION AT NULL POSITION PERPENDICULAR TO X-Y PLANE AND PARALLEL TO REFERENCE PLANE A WITHIN ± 0.5 MILIRADIAN
+Y
REFERENCE PLANE A
-Z
SCIENCE SCAN PLATFORM
SUPPORT SOLAR PANELS

10

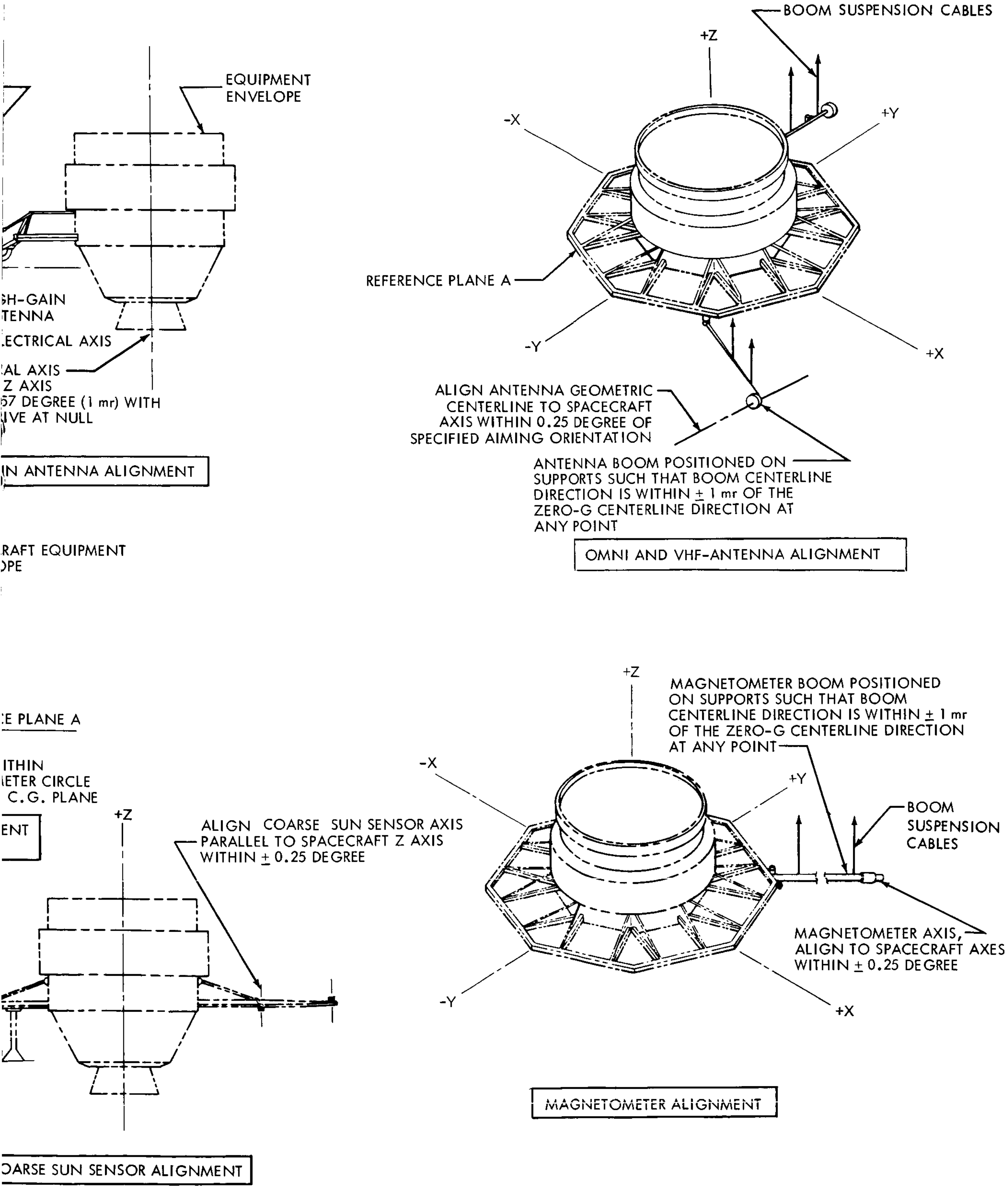


Figure 3.3-8: Voyager System Alignment Requirements

2

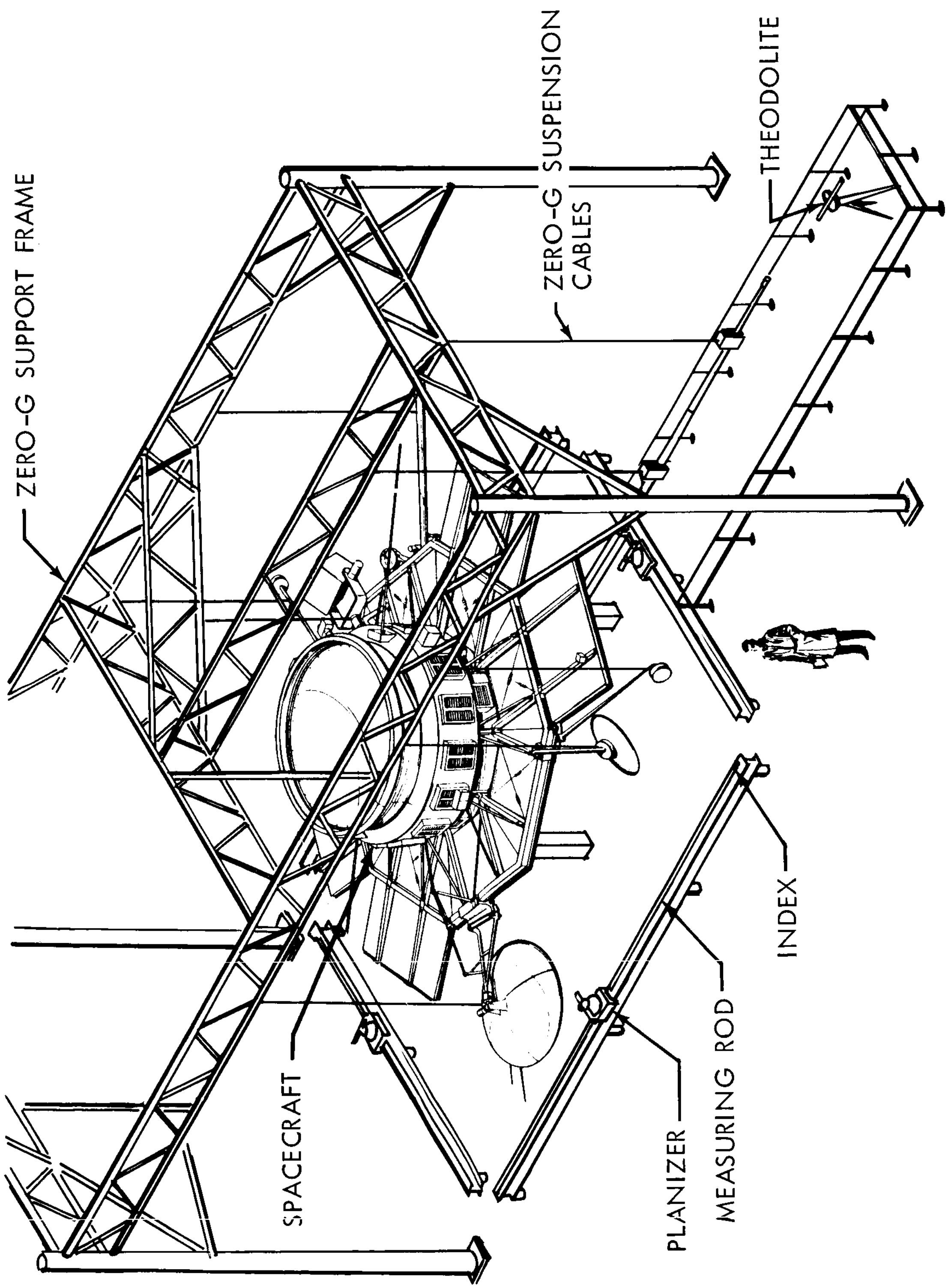


Figure 3.3-9: Spacecraft Alignment Equipment

Facilities requirements include a rigid base for the planizer bed and the alignment stand, and receptacles for light sources. The alignment stand accepts the spacecraft on the aft handling adapter (see Section 3.3.13).

The optical system used with the planizer will measure positions with respect to the spacecraft axes within ±0.005 inch and angles within ±1 milliradian (3.44 minutes of arc).

The alignment equipment consists of an alignment stand, four planizer rails approximately 30 feet long that surround the stand, and optical instruments, including telescopes, planizers, theodolites, and targets. Appendages are supported in the zero-gravity position during alignment by cables attached to an overhead framework.

3.3.5.3 Nose-Fairing Clearance-Measuring Equipment

This equipment is used during assembly and check-fit operations to verify proper clearances between the Planetary Vehicle and the nose fairing.

3.3.5.4 Orbit-Insertion-Motor Alignment Set

This equipment aligns the thrust-vector axis of the orbit-insertion motor to the Spacecraft/Planetary Vehicle center of gravity after live-motor or propulsion-module installation. It interfaces with the flight vehicle by a thrust-axis fixture. This fixture is mounted on the exterior of the motor nozzle and is aligned to the spacecraft axis targets.

The thrust axis of the orbit-insertion motor is measured to verify that it is parallel to the Planetary Vehicle Z-axis within ±1 milliradian (3.44 minutes of arc), and is coincident with the Z-axis within ±1/16 inch.

The orbit-insertion-motor alignment fixture is secured and aligned to the motor nozzle and is provided with two reference targets on each of two axes at 90-degree spacing. Two targets are installed on the spacecraft on each of two axes at 90-degree spacing. Portable optical instruments are used on two axes for alignment.

3.3.6 Test Stands and Fixtures

Test stands and fixtures are required to support all levels of assembly and testing. They vary in complexity from simple brackets used in attaching components to vibration tables, to fixtures that position the entire spacecraft in a variety of positions for a series of tests. Included as a part of test fixtures are the mechanical simulator devices that provide stimuli for the tests. The major stands and fixtures are described in the following paragraphs. Additional equipment items are listed in Table 3.3-1.

3.3.6.1 Spacecraft/Planetary Vehicle Assembly Stand

This stand supports the spacecraft structure while the equipment and the capsule are being installed. The stand interfaces with either the Planetary Vehicle Adapter or the aft handling adapter (Figure 3.3-10). Its capacity is approximately 30,000 pounds.

3.3.6.2 Inversion Stand

During inplant or Kennedy Space Center (KSC) handling, the spacecraft is in the same attitude as it is during the boost condition, with the solar panels pointed downward. At Boeing's Kent Space Center, the solar simulators radiate downward. It is, therefore, necessary to invert the spacecraft to point the solar panels toward the light banks for the free-mode and thermal-vacuum tests. Referring to Figure 3.3-11,

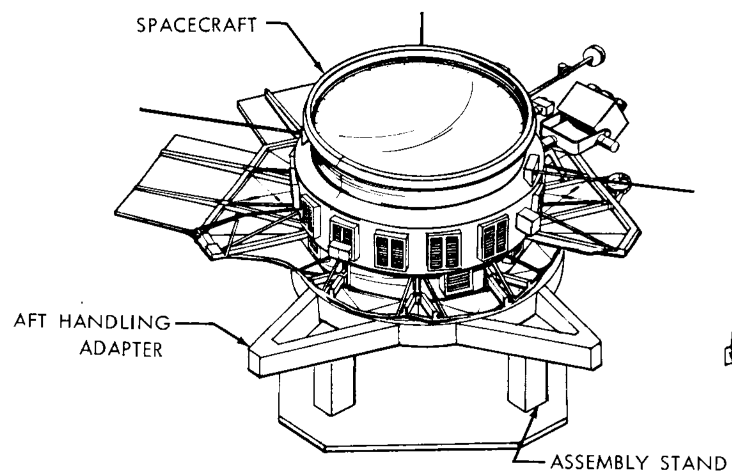


Figure 3.3-10: Spacecraft/Planetary Vehicle Assembly Stand

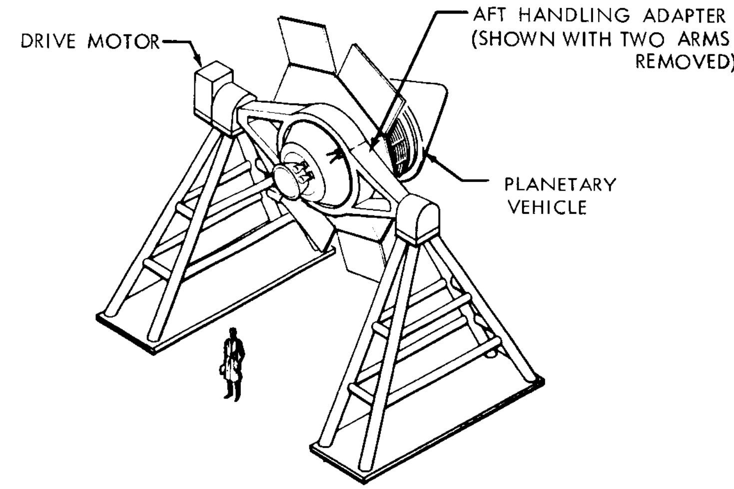


Figure 3.3-11: Inversion Stand

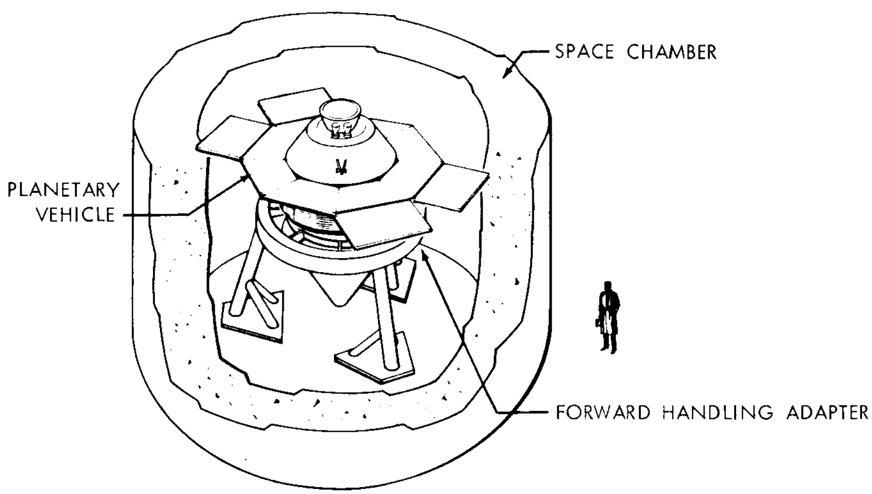


Figure 3.3-12: Free-Mode Test Stand

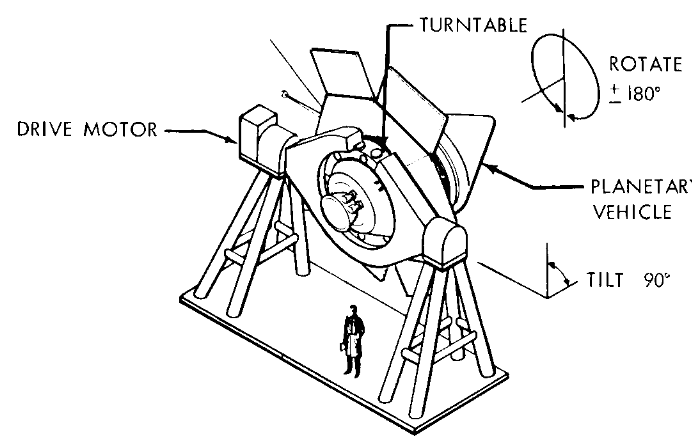


Figure 3.3-13: Spacecraft Positioner And Test Stand

inversion is accomplished by inserting two of the arms of the aft handling adapter into the inversion fixture. The inversion fixture pneumatic drive motor rotates the aft handling adapter with the spacecraft or the Planetary Vehicles through 180 degrees. Before inversion, the forward handling adapter is installed. After inversion, the spacecraft or Planetary Vehicle is lifted from the inversion fixture for installation in the free-mode test stand.

3.3.6.3 Free-Mode Test Stand

Free-mode testing is performed in the Boeing space chamber at Kent, Washington, using artificial light to activate the solar panels. The spacecraft or Planetary Vehicle is first placed in the inversion stand and inverted. It is then hoisted into the chamber, where it is installed on the free-mode test stand (see Figure 3.3-12). The upper end of the stand interfaces with the forward handling adapter. The aft handling adapter is removed. The free-mode test stand is made from nonmagnetic materials. It is insulated from the spacecraft to prevent heat flow by conduction and is made from materials that will not outgas in hard vacuum. This stand is also used for the thermal vacuum tests.

3.3.6.4 Spacecraft Positioner and Test Stand

This stand, shown in Figure 3.3-13 supports the spacecraft with or without the capsule for tests and adjustments. The stand positions the vehicle so that booms, antennas, and solar panels can be deployed, although not simultaneously, in all attitudes of the spacecraft. The Planetary Vehicle Adapter interfaces with the supporting ring on this stand. The stand is anchored to the facility floor and provides the capability of tilting the vehicle 90-degrees and rotating it about the Z-axis 180-degrees in each direction. Magnetic fields around the stand are minimized by aluminum construction and the use of pneumatic motors in the drive mechanisms. An identical stand is provided at Goldstone to support the spacecraft during testing.

3.3.6.5 Encapsulated Planetary Vehicle Assembly Stand

An assembly stand is provided at the explosive safe area (ESA). This stand supports the encapsulated Planetary Vehicle during nose-fairing installation. It is also used on the transporter during transit to the pad. It interfaces with the encapsulated Planetary Vehicle at the launch vehicle attachment locations and supports the vehicle in an upright position. It is constructed from aluminum to save weight and minimize magnetic effects. The stand has a capacity of 60,000 pounds and occupies a space approximately 21 feet in diameter and 4 feet high.

3.3.6.6 Propulsion Interaction Test Fixtures

Propulsion interaction tests are conducted to verify that the autopilot subsystem is capable of maintaining and controlling the spacecraft attitude during operation of portions of the propulsion subsystems.

A test fixture is provided for midcourse-correction and orbit-trim dynamic testing. This test requires that the Planetary Vehicle be placed in a space chamber and suspended in a manner allowing six degrees of freedom. In this test configuration, the midcourse motor is fired and the resultant behavior of the Planetary Vehicle and autopilot is monitored. The suspension system consists of a gimbal (permitting two degrees of rotational freedom) to which the Planetary Vehicle is attached. The gimbal is suspended in the space chamber by a cable that allows the third degree of rotational freedom and two degrees of translational freedom. The third degree of translational freedom is accomplished by routing the cable over a sheave located on the roof of the space chamber to a soft restraint.

A second test fixture is provided for making low frequency tests to determine the dynamic effects of orbit Inertion Motor firing on the attitude control system. This fixture provides a low-frequency soft mount for the spacecraft and a means to induce vibration into the motor nozzle to simulate motor firing.

3.3.6.7 Planetary Vehicle Nose-Fairing-Separation Test Fixture

This fixture provides the mechanism to pull the fairing halves clear of the Planetary Vehicle after separation.

3.3.6.8 Spacecraft Vibration-Test Fixtures

Fixtures are provided for vibration of both the spacecraft and the Planetary Vehicle. They are designed to rigidly fasten the specimens to the shake table, and provide no deflections that would influence test results.

3.3.6.9 Science Subsystem Test Fixtures

The Science Subsystem test fixture is a structure similar to the spacecraft structure in overall size and shape. The fixture is finished to flight hardware tolerances at the Science Subsystem attachment points. Actuation mechanisms are included in the fixture to support the portions of the Science Subsystem at which actuation is required and which are not included in the subsystem. Portions of the fixture can be removed so that detail tests of parts of the subsystem can be performed in specialized laboratories.

3.3.6.10 Zero-g Test Equipment

The spacecraft contains booms, antennas, and panels that normally actuate in a zero-g environment. To test this equipment in the laboratory, Earth gravity must be counteracted by the application of forces with equal magnitude, but opposite direction. Methods of gravity counteraction are generally: (1) support of the equipment from a long overhead cable, allowing the test item to deploy in the horizontal plane, or (2) support of the deploying item on soft support stands that are free to roll on a smooth floor.

3.3.6.11 Capsule/Canister Separation Test Fixture

During this test, the canister separates in sections from the capsule and the capsule from the spacecraft. The spacecraft/Planetary Vehicle assembly stand is used for this test. With the Planetary Vehicle in an upright, nozzle-down position, a system of lines, pulleys, and counterweights is attached to the canister. The lines are placed in tension by the centerweights to simulate zero-g conditions. The system does not interfere with the separation test and prevents the canister sections from damaging the spacecraft or adjacent equipment after the test. The capsule is separated from the spacecraft in a similar manner.

3.3.6.12 Subsystem Type-Approval Test Fixtures and Equipment

The subsystem type-approval test fixtures and equipment consist of hardware items designed to subject one or more of the flight subsystems to ground environment tests such as drop, transport vibration, and humidity. Because each flight subsystem is unique, the fixtures are tailored to fit the requirements of each subsystem to be tested.

3.3.6.13 Subsystem Launch and Flight Environment Type-Approval Test Fixtures

This category includes a number of fixtures that are used to test the flight subsystems for their reaction to the launch and flight environment. Each subsystem has a fixture to support and load the subsystem in a manner comparable to that which the subsystem experiences during launch or flight. Typical of the loads applied by these fixtures are shock (boost, pyrotechnic, and thermal), acceleration, vibration, vacuum, and magnetism. Type-approval testing, in some cases, requires that the equipment be tested to loadings in excess of those experienced in flight. The fixtures are capable of meeting this requirement.

3.3.6.14 Dynamic Loads Test Adapter

This adapter holds the spacecraft on a vibration stand for the dynamic loads test. It consists of a dummy forward section of the Saturn IVB with provisions for mounting it on a vibration table. The spacecraft is held in place by the Planetary Vehicle Adapter.

3.3.6.15 Magnetic-Mapping Equipment

Magnetic mapping is accomplished by the same basic method as the Mariner spacecraft.

The equipment shown in Figure 3.3-14 determines the magnetic field, resulting from the spacecraft,at the flight magnetometer location. The spacecraft can be positioned at any set of points on a circular

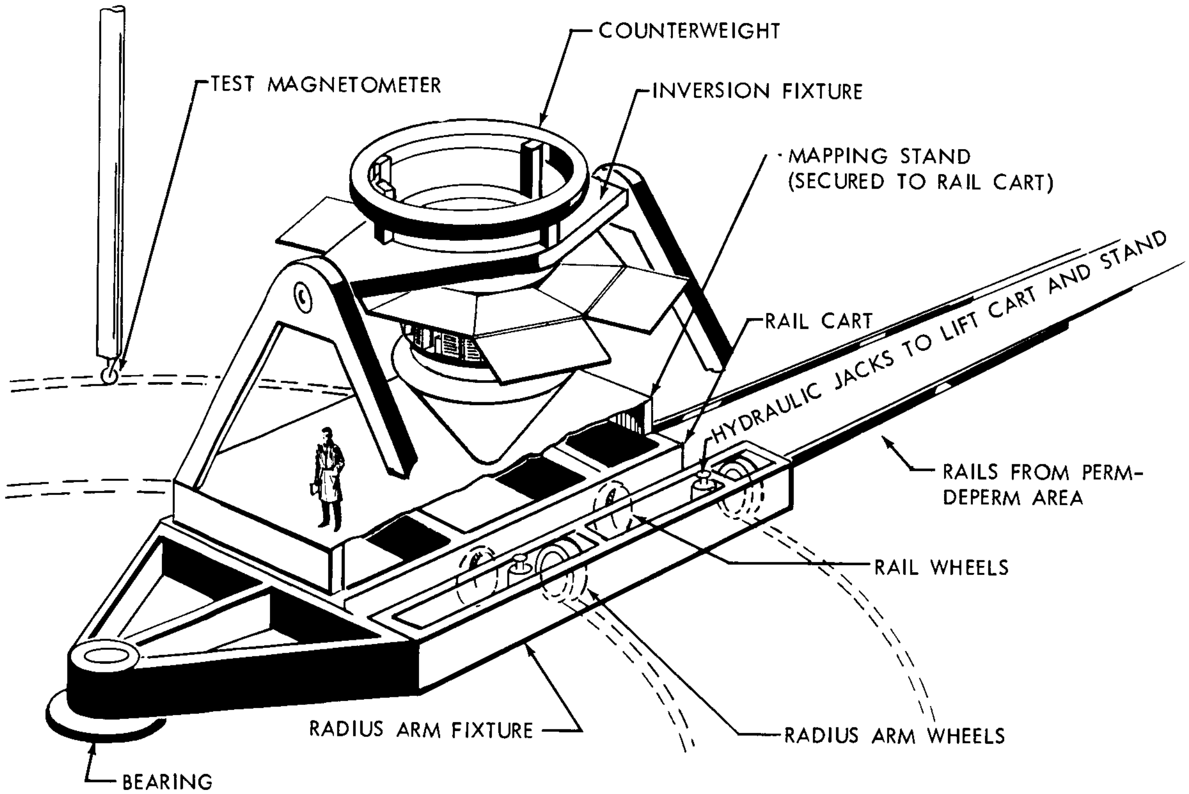


Figure 3.3-14: Magnetic Mapping Fixture

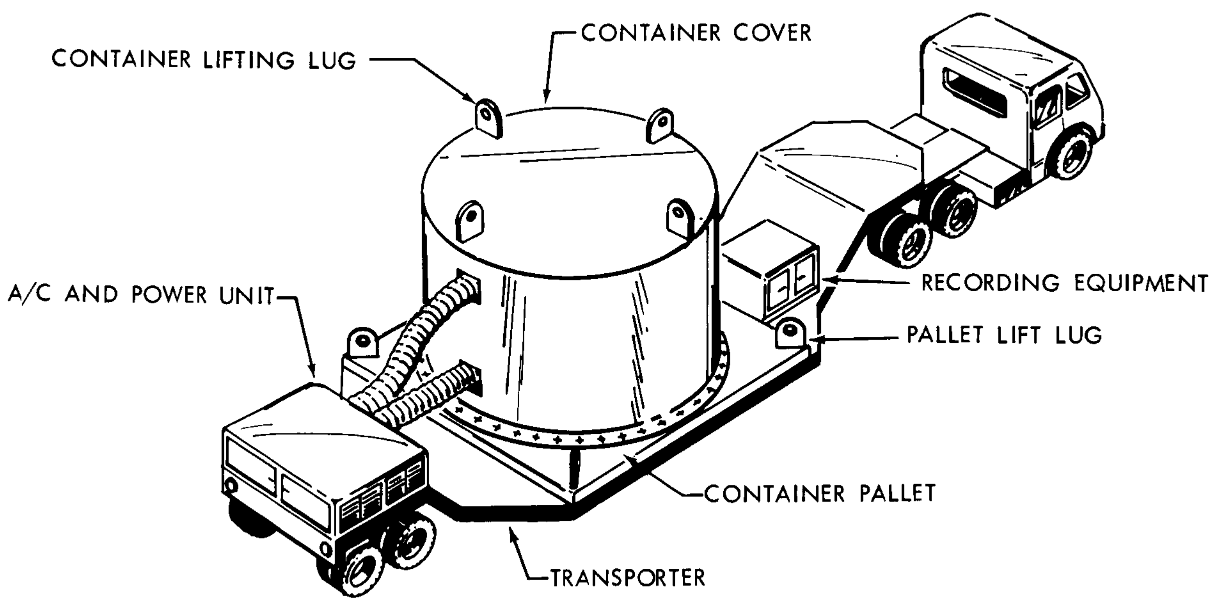


Figure 3.3-15: Spacecraft Transporter

track whose radius is equal to the length of the magnetometer boom. In addition, the spacecraft can be rotated 180 degrees around the magnetometer boom axis. The magnetometer boom is removed for the mapping. The equipment allows antennas, solar panels, etc., to be in their deployed positions during mapping. The equipment uses pneumatic drive and is of entirely nonmagnetic construction. Instrumentation for spacecraft or Planetary Vehicle mapping consists of a triaxial mapping magnetometer, a triaxial reference magnetometer, and control equipment including indicators and automatic data equipment.

3.3.6.16 Perm-Deperm Equipment

The perm-deperm equipment includes a test stand, a coil assembly, and power control system instrumentation. The perm-deperm stand is a three-axis gimbal device capable of simultaneously and randomly rotating the Planetary Vehicle about the X, Y, and Z axes. All mechanisms are in launch position and no external connections are required. Nonmagnetic construction is used to maximize deperming effectiveness.

The stand is secured to the facility floor and the Planetary Vehicle is mounted by crane.

The coil assembly consists of two 60-foot-diameter coils 30 feet apart, and an aluminum supporting structure. The stand locates the Planetary Vehicle in the center of the coil field.

Approximately 100 kilowatts of facility power are used to provide a controlled d.c. perming field of 25 oersteds and a controlled a.c. deperming field of 50 oersteds. The maximum rates of change of the perm-deperm fields are limited by the equipment design to protect spacecraft electrical circuits from damage.

3.3.6.17 Acoustic Test Fixture

This fixture is used with the spacecraft vibration test fixture to provide a soft suspension for acoustic tests.

3.3.7 Transportation Equipment

Transportation equipment consists of items used in over-the-road transportation of Voyager system elements. Included are the spacecraft transporter, orbit-injection-motor transporter, OSE remote site transporters, solar-panel shipping trailers, loading ramps, and tie-downs.

3.3.7.1 Spacecraft Transporter

The spacecraft transporter is required for spacecraft highway transportation. The unit is also used for moving between clean-room facilities at Seattle or Eastern Test Range (ETR), moving to and from the aircraft loading area, and between the ESA and the launch pad.

The transporter is a special trailer and 10-wheel tractor, as shown in Figure 3.3-15. The trailer bed accommodates the spacecraft shipping container, the transportation air conditioning unit, and the transportation recording system. A storage compartment for loading equipment is provided.

The transporter interfaces with the spacecraft or Planetary Vehicle shipping container. The transporter load capacity is 40,000 pounds, to accommodate an encapsulated Planetary Vehicle on its assembly stand and the environmental control unit. The suspension system limits spacecraft/Planetary Vehicle acceleration to 5 g's vertically and 2 g's horizontally. The transporter will negotiate a 5-percent grade with a 120-percent proof load aboard.

Testing includes road tests for shock, vibration, and stability, including simulated wind loading tests; environmental tests with equipment operating; and proof load and performance.

3.3.7.2 OSE Remote Site Transporters

OSE items are required for operations at remote sites and are transported using conventional trucking equipment. Enclosed trailers or flatbed trailers are provided to move test equipment, stands, and handling equipment. Suspension rates and transportation environment are controlled and monitored for critical equipment.

3.3.7.3 Solar-Panel Shipping Trailer

The method of shipping solar panels is identical to that successfully employed by JPL on Mariner C. The solar panels are mounted on a handling frame and sealed in a plastic bag purged with clean, dry nitrogen. The handling frame with solar panel is then installed in a transportation van that attentuates shock to acceptable levels. Boeing will study the possibility of modifying existing JPL solar-panel transportation vans for use on Voyager.

3.3.7.4 Transportation Recording Equipment

A rack is provided that contains equipment to monitor and record temperature, humidity, acceleration, and magnetic environment of the spacecraft and its components during transport. Status indicators of the transportation power system and environmental equipment, and alarm functions for critical parameters, are included in a display unit that can be located either remotely or in the rack. The unit is skid mounted and contained in a weather-proof housing. The equipment rack is secured to the spacecraft transporter. Sensors are mounted on or in the shipping container. Power (110-volt, single-phase, 60-cycle) is obtained from the transporter unit or from facilities when the transporter unit is inoperative. The display unit is mounted in the transporter cab during movement.

Temperature is monitored at two points in the container, four points in the air conditioning unit, and one point in the air conditioning unit generator. All temperatures are recorded on a strip-chart recorder. Container humidity is monitored by redundant sensors and automatically recorded. The container accelerations are measured in three axes by three separate unidirectional transducers and are recorded. Two redundant, nondirectional, magnetic-field transducers are used with a recorder during transportation. An audible alarm is provided before operating limits of temperature and humidity are reached. The instruments are individually calibrated.

3.3.8 Safety Equipment

All system level AHSE affords personnel protection and provides for the safe handling of flight hardware. This feature is incorporated in the AHSE by selecting conservative designs, considering operator error sources, eliminating risky operations, and by following established procedures and criteria. Special safety equipment to further reduce the chance of equipment and personnel injury are included.

Hazardous tests areas are isolated by blast barriers and revetments. Personnel conducting fueling operations will wear protective clothing and will be provided with self-contained breathing apparatus. The fueling facility contains complete personnel decontamination equipment, including an eye-wash fountain and personnel emergency shower.

3.3.9 Work Platform Sets and Access Equipment

Work platform sets and access equipment are provided for personnel access to the spacecraft during assembly and test operations (see Table 3.3-1).

3.3.10 Dollies

Dollies (Figure 3.3-16 are provided for moving and positioning components, assemblies, spacecraft, and Planetary Vehicles within assembly and test areas, are designed for shop or laboratory use, and are not roadable.

All dollies have safety locks, resilient bumpers, configurations resistant to overturning, conductive finishes, conductive tires, and grounding connections. Motors and controls on dollies used in hazardous areas are equipped with explosion-proof housings.

3.3.11 Protective Covers

Protective covers are provided to maintain a clean-room environment during transit of all flight hardware and during periods of inactivity.

Protective covers used on or near flight equipment are made of electrically conductive materials, such as Velostat, and are similar to those successfully used by Boeing on the Lunar Orbiter program. Provisions are made for grounding the covers. Table 3.3-1 lists the protective covers required.

3.3.12 Shipping Containers

Shipping and handling containers are used in shipment of the spacecraft and its components among manufacturer's plants, and between manufacturer's plants and KSC. These containers are designed to protect their

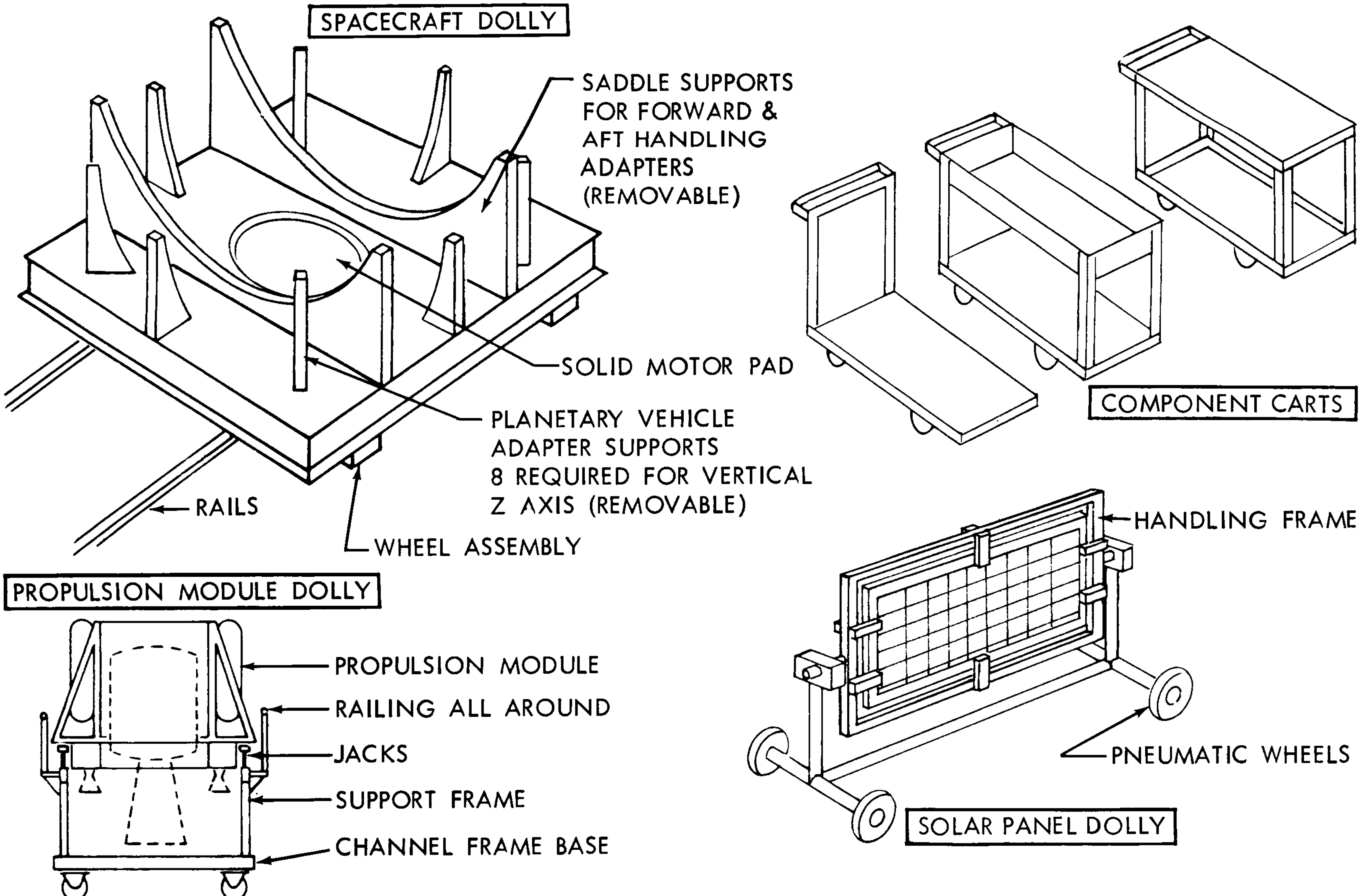


Figure 3.3-16: Dollies And Carts

contents from natural and induced environments and to facilitate handling during shipment. They are designed for air and highway transportability. The Planetary Vehicle shipping container is compatible with the B-377PG-2 aircraft; all other containers are compatible with commercial cargo aircraft. When required, the containers provide for recording the following environments to which their contents are subjected: temperature, pressure, humidity, shock, vibration, and magnetic field strength. The containers are designed to permit airflow from the environmental conditioning unit such that temperature, pressure, and humidity requirements are met at all critical points on the contents.

The spacecraft shipping container (Figure 3.3-17) is used for local transportation of the spacecraft. The spacecraft is mounted on the aft handling adapter (Section 3.3.13), which in turn is rigidly fastened to the base platform of the container. Four lifting lugs are located on the platforms for lifting the entire package. Lifting lugs are also provided on the container cover. Air conditioning ports are located on the container cover, and the container is insulated. An adapter is provided for inplant handling of the Planetary Vehicle. This adapter is a cylindrical section bolted to the cover of the spacecraft shipping container. The length of the cylindrical section is equal to the height of the capsule plus clearance for the Planetary Vehicle when installed under the cover.

The Planetary Vehicle shipping container, shown in Figure 3.3-18, is used to ship the Planetary Vehicle from Seattle to ETR. The container is a horizontal cylinder and is separable at the horizontal centerline. The lower half has rails for support and attachment to the spacecraft and the B-377PG-2 aircraft. The forward and aft handling adapters are

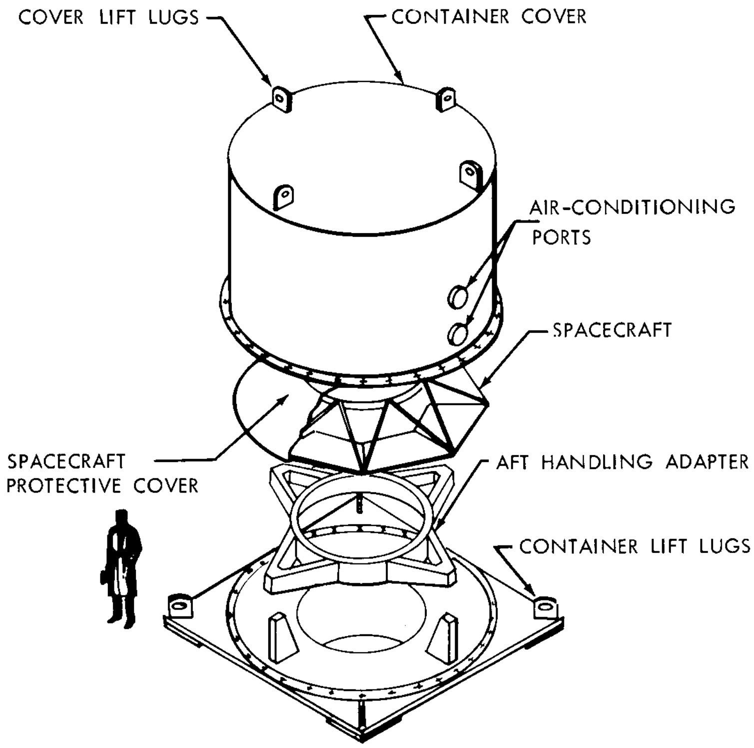


Figure 3.3-17: Spacecraft Shipping Container

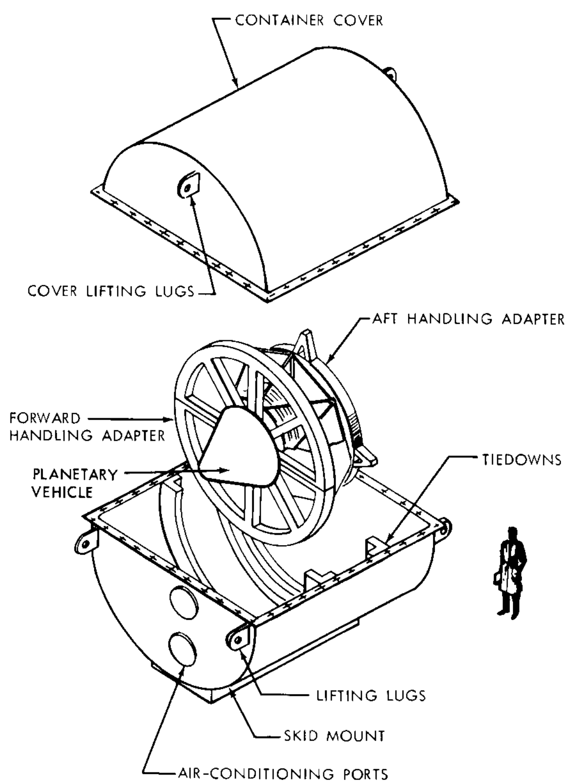


Figure 3.3-18: Planetary Vehicle Shipping Container — Air Transport

used to lift the Planetary Vehicle in the horizontal attitude and install it in the lower half of the container. The upper half of the container provides the closure for the vehicle. The ground transporter provides rails to load the Planetary Vehicle and shipping container in the B-377PG-2. The container is disassembled to permit its return to Seattle by either truck or commercial cargo aircraft. The orbit-insertion-motor shipping container, provided by the motor subcontractor, is used to ship motors from the manufacturer's plant to KSC.

Subsystem components carried as spares or transported separately from the spacecraft are provided with shipping containers. These containers are equipped with humidity indicators and pressure-relief ports. Air passing through the relief ports is routed through a dessicant inside the container to reduce humidity. The louvers are mounted in the container by fastening the frame to mounting brackets on the box. Other subsystem and component containers are listed in Table 3.3-1.

Shipping containers are subjected to acceptance testing with simulated contents. Containers for which atmospheric environmental control is required are tested, with their environmental control units, in an altitude chamber simulating cargo-aircraft pressure and temperature. Since all containers are used (to some extent) in over-the-road transportation, they are also tested for shock and vibration transmissibility under road-transportation conditions.

3.3.13 Lifting and Handling Devices

Lifting and handling devices are provided and are listed in Table 3.3-1. Standard equipment is used where possible. Special lifting hardware is provided where standard slings and riggings cannot be used.

Except for handling during free-mode and thermal-vacuum tests, all handling of the spacecraft and Planetary Vehicle is done by using the aft handling adapter (Figure 3.3-18). This handling adapter adapts the Planetary Vehicle or spacecraft to all test fixtures (magnetic-mapping fixture, perm-deperm fixture, and inversion fixture) the spacecraft shipping container, and the lifting and handling devices used for hoisting.

The spacecraft forward handling adapter provides capability for lifting and handling the Planetary Vehicle during the free-mode and thermal-vacuum tests. It is a ring that fastens to the Spacecraft Bus at the capsule/spacecraft interface. Three lifting lugs are provided for sling attachment.

The spacecraft sling is a three-cable sling used with handling adapter rings to hoist the spacecraft. The forward handling adapter (attached to the Spacecraft Bus) positions the Spacecraft Bus on the handling adapter or the Planetary Vehicle (flight) Adapter. With the capsule installed, the load path is to the three arms on the aft handling adapter or to the aft ring, which distributes the load to the Planetary Vehicle Adapter.

The encapsulated Planetary Vehicle lifting equipment consists of a distribution ring structure that takes two-point loads from a sling (composed of a simple beam and cables, providing single-point pickup).

Another item of OSE lifting equipment is the spacecraft shipping container sling, which consists of a beam and spreader cables that pick up four lugs on the shipping container base. Attachment points are also provided on this beam for lifting the cover of the shipping container. Additional OSE lifting equipment includes the aircraft loading and unloading equipment. This equipment facilitates loading the Planetary Vehicle shipping container into the transport aircraft.

3.3.14 Servicing Equipment (also see Section 4.3)

Servicing equipment is provided to supply conditioned fluid commodities to the spacecraft; flush, purge, and dry the spacecraft and ground equipment; decontaminate the internal surfaces of the spacecraft reaction control and thrust vector control (TVC) fluid system hardware; and perform functional or qualitative tests on the spacecraft equipment. Table 3.3-1 lists the individual items within this equipment category.

3.3.14.1 Biological Decontamination Equipment

The spacecraft requires the following services from ground equipment to comply with planetary quarantine allocations:

1) Internal (wetted) surface decontamination of the reaction control and the TVC fluid containing hardware.
2) Decontamination of TVC freon.
3) Decontamination of TVC pressurization and reaction control system nitrogen.
4) Decontamination of helium used during spacecraft systems' leak checks.

The above functions are satisfied by the decontamination equipment employing biological filtering within the Servicing Unit (ref. Paragraph 4.3.1.5) and ethylene oxide as a steriant for both spacecraft and ground equipment internal surface treatment followed by post-treatment sterile nitrogen purge. Supply gases, including helium and nitrogen, are decontaminated by filtration where low flow rates and quantities are required, nitrogen is decontaminated by heating and subsequent cooling within the ground unit. Trade studies in Phase 1B are necessary to evaluate the hardware availability, performance, and cost effectivity of heat versus filtration decontamination methods for gases.

Basic units of the decontamination equipment consist of gas supply units (nitrogen and helium,) a nitrogen gas sterilizer and cooler, gas biological filtration units, gas distribution systems and controls and an ethylene oxide treatment unit. These equipments interface with the spacecraft directly or with the various propulsion and reaction control subsystem service or test units described in Section 4.0. Decontaminated or standard gas quantities are delivered to these interfacing units as required for ultimate usage aboard the spacecraft. Decontaminated connectors are provided for delivering all commodities to the spacecraft or other interfacing ground equipment. Provisions are made within this equipment to establish and maintain decontaminated lines and components for those circuits delivering decontaminated fluids and gases.

3.3.14.2 Particle-Contamination Detector

Prior to encapsulation, certain spacecraft surface areas are examined for presence of particles 4 mils in diameter or larger. The portable particle-contamination detector accomplishes this by providing an intense source of white light that is focused almost parallel to the surface being examined. Particles on the surface will cause light scattering or a partial light diffusion. Reflected light from a particle-contaminated surface is compared with light from a standard smooth surface of the same finish by means of a calibrated photoelectric cell and meter.

3.3.15 Installation Kits and Assembly Equipment

The necessity of replacing subsystems and major components during system testing at Kent and ETR requires that assembly equipment for the entire spacecraft be available at both locations. Assembly fixtures, gages, and tooling are made available in the form of kits (see Table 3.3-1) at the central system test facility at Kent and the assembly and checkout facility at KSC.

3.3.16 Environmental Control Equipment

Environmental control equipment is provided to maintain the spacecraft and Planetary Vehicle at the required temperature and humidity levels during testing, transportation, and storage.

3.3.16.1 Spacecraft Transportation Air Conditioning Unit

The air conditioning unit provides a means of controlling the environment to the spacecraft shipping container during moving operations outside of clean areas. It is mounted on the spacecraft transporter trailer. Flexible ducting transfers the conditioned air from the unit to the shipping container. The temperature of the air entering the container is controllable between 40 and 80^{o}F and is provided at a rate of 1500 scfm. The delivered air is filtered by absolute filters of 99.95 percent efficiency for 0.3-micron particles. The unit has a cooling capacity of 60,000 Btu per hour and a heating capacity of 20,000 Btu per hour (6 kilowatts).

The air conditioning unit is a skid-mounted assembly powered by a 7-kilowatt motor generator providing 440-volt, three-phase, 60-cycle power. A switching unit is provided so that the unit may be used with ground or aircraft power. A 110-volt, single-phase, 60-cycle circuit is provided to supply power to the transportation recording equipment.

3.3.16.2 Encapsulated Planetary Vehicle Environmental Control Unit

This unit provides a supply of filtered, temperature- and humidity-controlled air to the vehicle when encapsulated in the nose fairing.

The nominal cooling capacity and performance of this unit are identical to that used in the transportation air conditioner. Power is obtained directly from the facility. Absolute filters are provided. The unit is skid mounted and is suitable for clean room use. The electrical system is explosion-proof.

3.3.16.3 System Test Cooling Unit

This portable unit provides cooling air for the spacecraft electronic and power components during system-level testing. The cooling air is pumped through four 4-inch-diameter flexible ducts positioned to direct the air onto the louvers of the spacecraft. This air is provided at a rate of 500 cfm at a pressure of 10 inches of water. The temperature is maintained between 45°F and 75°F with a maximum relative humidity of 50 percent.

<u>3.3.17 Ancillary Equipment</u>

In addition to primary AHSE, ancillary equipment is provided to support testing operations. This equipment is as follows:

1) Solar panel dummies;

2) Dummy capsule;
3) Orbit-insertion-motor simulator;
4) Solar simulator;
5) Service arm simulator;
6) Solar panel illuminator;
7) OSE magnetic evaluation equipment;
8) Subsystem magnetic-mapping equipment;
9) Launch environment vibration and acoustic test equipment;
10) Pyrotechnic shock test instrumentation;
11) Container washdown unit;
12) Reflectance test equipment;
13) Antenna pattern test equipment;
14) Solar panel deployment aids;
15) OSE environmental vibration, shock, temperature, and humidity test equipment;
16) Subsystem module support stands;
17) Structural test model fixtures.

3.4 SPECIAL TEST FACILITIES

This section discusses the characteristics of special test facilities required for the Voyager at Kent and KSC.

3.4.1 Summary

Special test facilities are those required to support the test and operation requirements of the Voyager program. This section outlines special test facility requirements.

Special test facilities are required for: design verification testing of selected components and subsystems, type-approval testing, flight-acceptance testing, and prelaunch tests.

The Boeing Company has constructed extensive test facilities at its Kent Space Center. These facilities, existing and planned, will satisfy a substantial portion of Voyager's major requirements.

3.4.2 Applicable Documents

The documents used in defining special test facility requirements are identified in Section 2.1.

3.4.3 Design Constraints and Requirements

Special test facility design requirements and constraints are developed simultaneously with those developed in Section 3.3.3 for AHSE.

3.4.3.1 Compliance with System Specifications

"Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification for" was used as a guideline for the special test facilities. Described below are examples of these special test facilities.

Thermal Vacuum Simulator--Paragraphs 4.3.2(1) and 4.3.3(1) of the general specification require: "...environmental tests shall include space simulation test in which the Spacecraft is exposed to a thermal-vacuum profile, simulating...the space and boost environment to which it will be exposed."

A large space chamber and associated equipment to support this requirement is available at the Kent Space Center. The space chamber (Section 3.4.5) provides the full range of pressure, temperature and solar simulation capabilities required for Voyager space simulation testing.

Environmental Test Laboratories--Section 4.0 of the general specification requires tests and verification of all parts, components, subassemblies, subsystems, and systems. In addition to the specific special test facilities described herein, test facilities are available throughout The Boeing Company to perform the environmental tests on components.

Propulsion Interaction Tests--Paragraphs 4.2.1.3 and 4.3.2.16 of the general specification requires: "Propulsion Interaction Tests shall be conducted to verify that the autopilot subsystem is capable of maintaining and controlling the Spacecraft attitude during the operation of the propulsion subsystem, and to verify that the dynamic properties of the Spacecraft structure do not have a harmful effect on autopilot performance."

Propulsion interaction testing facility requirements can be satisfied as delineated in Section 3.3.6.6 and 3.4.5.5.

Additional Facilities--Additional major impact on the special test facilities are the cleanliness requirements and the safety requirements. Provisions to meet these requirements are provided in all test areas where operations are performed on the flight hardware and the proof test model (PTM).

3.4.3.2 Exceptions

No exceptions are taken to the general specification in respect to defining facility requirements.

3.4.3.3 Derived Requirements

Section 2.0 describes the expansion of the functional flow diagrams used to develop the AHSE. The facility requirements are also derived from the expansion of the same functional flows (see Figure 2.4-1 for an example).

The facility requirements are identified in Table 3.4-1.

Table 3.4-1: FACILITY REQUIREMENTS

<u>Type Approval and Development Testing--Seattle</u>
- * Launch environment simulation test facility
- * Component and assembly magnetic test facility
- * Antenna test range facility
- * Space simulation test facility

<u>Flight Acceptance Testing--Seattle</u>
- * Space simulation test facility (same as FAT)
- * Electromagnetic interference (EMI) test facility
- * Magnetic test facility
- * Launch environment simulation test facility
- * Environment controlled clean assembly test areas

<u>Facility Requirements at KSC</u>
- * Receiving facilities (air locks and cleaning areas)
- * Explosive safe areas
- * Magnetic test facility
- Decontamination facility
- * Environment controlled clean assembly and test areas
- * Mobile launch tower (modified)

Note: The asterisks indicate facility requirements supported by a requirement in JPL documentation.

3.4.4 Trade Studies

The facilities baseline was established and evaluated by process of trade study. The significant results are:

1) Location of facilities in Seattle--Seattle operations use the Boeing Space Center at Kent, which will be expanded to meet Voyager program requirements.

2) Receiving and inspection requirements--A two-station receiving and inspection area is required to ensure cleanliness and provide the area for handling the spacecraft with its associated transportation and shipping equipment.

3) The size of the Seattle assembly and test building--The size is established by consideration of the area needed for a wide variety of tests with as many as five spacecraft in this area at one time, and that the capsules (or capsule PTM's) will enter into tests in this area. Auxiliary areas have been sized on the basis of test, test support, and test control needs in relation to the activities described above. Other major trade studies concern the following special test facility functions:

   a) Magnetic testing;

   b) Free-mode tests;

   c) Propulsion interaction tests;

   d) Other tests such as antenna range, radio-frequency interference, acoustical, and vibration tests.

3.4.5 Seattle Facilities

Voyager processing and testing operations in the Seattle area require the following facilities:

1) An assembly and test building for assembly and testing of the spacecraft and Planetary Vehicles;
2) Magnetic test facilities, for magnetic measurement tests and the conditioning of test articles before and after testing;
3) Space simulation test facility for tests in simulated space environment conditions;
4) Launch environment simulation facility in which the flight hardware configurations are subjected to acoustic and vibration excitation;
5) Ancillary facilities for propulsion subsystem tests, separation test and antenna test range.

Table 3.4-2 lists the types of test operations performed in each of the Seattle facilities. Figure 3.4-1 is a layout of the assembly and test building at the Boeing space research laboratories at Kent. Detailed descriptions of the various special test facilities follow.

3.4.5.1 Assembly and Test Building

The assembly and test building is an integrated structure, housing areas ranging through receipt of parts and materials, fabrication, assembly, and testing.

The receiving, inspection, and shipping operation area includes:
1) An air lock to aid in maintaining cleanliness of the adjoining major test area;
2) Space in which the large hardware items can be received, their containers cleaned and opened, and container contents inspected or prepared for packaging;

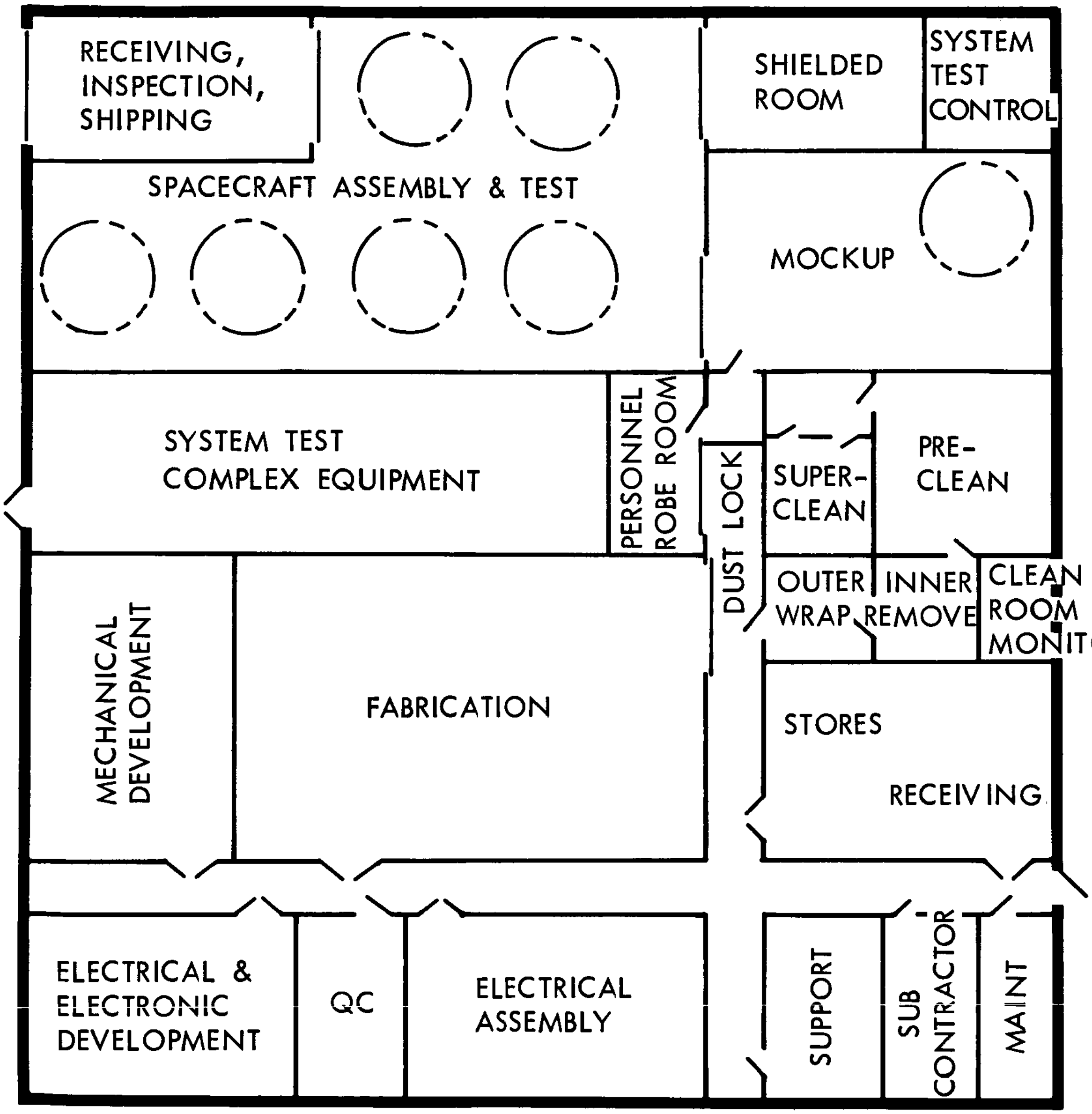


Figure 3.4-1: Assembly And Test Building — Boeing Space Research Laboratories

Table 3.4-2: VOYAGER SEATTLE FACILITY REQUIREMENTS

(Special Test Facilities)

ASSEMBLY AND TEST BUILDING

Initial power application
Subsystem tests
Intersubsystem tests
Telemetry channel calibration
Spacecraft system testing
Planetary Vehicle system testing
Checkfits (Spacecraft to nose fairing)
Interface tests
EMI tests
Dummy run countdown and launch
Alignment of Spacecraft elements
Weight/balance checks
Science Payload integration
Encapsulated Planetary Vehicle assembly tests (rf loop, purge, fit)
Failure mode and logic tests
Relay link tests

Magnetic Test Facilities

System magnetic test facility
1) Spacecraft or Planetary Vehicle Perm-Deperm
2) Spacecraft or Planetary Vehicle Magnetic Mapping

Component and Assembly Magnetic test facility
1) Subsystem, subassembly, component, part, etc., Perm-Deperm operations
2) Subsystem, subassembly, etc., magnetic tests

Tulalip Facility

Midcourse propulsion firing test
Separation tests

Space Simulation Chamber Building

Spacecraft thermal-vacuum test
Planetary Vehicle thermal-vacuum test
Simulated mission test
Free-mode test

Launch Environment Test Facility

Boost environment simulation test

Propulsion Interaction Test Facility

Midcourse propulsion interaction test

Antenna Test Range

Antenna test
Rf pattern test

3) Rails for article transfer into the test area by means of the spacecraft dolly. The area is required to meet Federal Standard 209, Class-100,000 standards, which is typical of all areas to be discussed in the following facility descriptions, unless specific exception is made.

The receiving, inspection and shipping area (RISA) is 40 by 80 feet, divided into two work stations. The innermost station has a grating floor over a depressed drain, where washdown equipment (an OSE item) is used to clean incoming containers. An overhead handling system of 20-ton capacity, with a maximum hook height of 60 feet is required. The facility provides hot and cold water to the washdown OSE.

The handling, assembly, testing, and checkout area includes:

1) A 20,000-square-feet assembly and test area, served by an overhead handling system of 20-ton capacity, with maximum hook height of 60 feet. This is a clean area. The floor has removable modules for access to space below for routing of test cabling between test articles and the STC. A shielded area is provided for EMI testing.

2) The test control room area provides for installation and use of the STC, LCE, and MDE in conducting the tests. Each control room has 3000 square feet of floor area, and environment control for comfort conditioning for personnel. Windows are provided to enable viewing of operations in the test areas. The facility needs a total area of 15,600 square feet.

3.4.5.2 Magnetic Test Facilities

The magnetic test facilities provide for operations necessary to: receive, inspect, and ship test articles (similar to the corresponding areas described in Section 3.4.5.1); magnetically treat test articles; magnetically map test articles; and control and support facility and test equipment.

The major features of the magnetic test facility are the two primary test buildings, the control buildings, and the support area, as described below.

Spacecraft/Planetary Vehicle Test Building--The spacecraft/Planetary Vehicle test building includes an RISA, a perm-deperm room, and a magnetic mapping room. The perm-deperm room is between the other two rooms and houses two transfer stations; a 60-foot-diameter Helmholtz coil; OSE perm-deperm fixture; transfer rails on 10-foot gage into the RISA; transfer rails on 20-foot gage into the mapping room; and an overhead handling system with 20-ton capacity and a maximum hook height of 60 feet. Maximum power delivered to the coils is 100 kilowatts. The perm-deperm room has 7000 square feet of floor area.

The magnetic mapping room requires 14,000 square feet of floor area and a clear height of 60 feet.

The buildings housing magnetic test and preparation operations are located in a magnetically stable area, remote from heavy electrical equipment or automobile-size masses of iron and steel, particularly those that move. The magnetic field gradient in the completed facility is less than 100 gamma per foot.

Component and Assembly Magnetic Test Facility--The component and assembly magnetic test facility is similar to the spacecraft/Planetary Vehicle test building in function, but a smaller building is required because the articles to be tested (components, subassemblies, assemblies, parts, and materials) are smaller.

The RISA is replaced with a 1200-square-foot room that provides for container cleaning, packing and unpacking, storage space, and a robing room.

The perm-deperm area requires 5000 square feet of space and a 10-ton overhead handling system with a maximum hook height of 45 feet. Rails must extend from the perm-deperm area into the magnetic test room to support the magnetic test fixtures.

The magnetic test room requires 2500 square feet of space, with a clear height of 50 feet.

The control room serves both magnetic test areas and contains power control and instrumentation required for test control. The required area is 800 square feet and is located not less than 350 feet from either of the test buildings.

The support areas and other facility features include: environment control equipment for the test buildings; transformer equipment; spacecraft or Planetary Vehicle cooling gas conditioning equipment; power SSTE; and nonmagnetic ducting for commodities/power/communication connections to the test and preparation areas. Buildings and facility items in the test areas described above must be made of nonmagnetic materials in all possible cases. No installed equipment may be closer than 350 feet to the test buildings. Commands, responses, monitoring and spacecraft d.c. power delivery are via specially shielded coaxial cabling.

3.4.5.3 Space Simulation Test Facility

Activities to be performed in this facility are thermal-vacuum testing of the spacecraft (or Planetary Vehicle), simulated mission tests, and free-mode tests. The building has, in addition to the large chamber, two 10-foot-diameter space simulation chambers, a vibration stand, and space for STC equipment. This facility is equipped with a RISA as described in Section 3.4.5.1.

The significant feature in this facility for system testing is the 39-foot-diameter by 50-foot-high space simulation chamber. The chamber can maintain pressures down to 1 x 10^{-9} torr and temperatures of -300 to +280°F. Solar simulation equipment is also provided to allow free-mode testing in this facility.

The room illustrated in Figure 3.4-2 is 100 by 100 feet in plan, equipped with a bridge crane with maximum hook height of 86 feet, and has an access door opening (minimum) 26 feet wide by 40 feet high.

3.4.5.4 Launch Environment Simulation Test Facility

This facility provides for testing of spacecraft or a Planetary Vehicle in simulated boost and flight environment conditions of sound and vibration.

Interfaces include facility handling system/OSE handling devices, support points/OSE holding fixtures, power/OSE, and routeways/OSE interconnecting cabling.

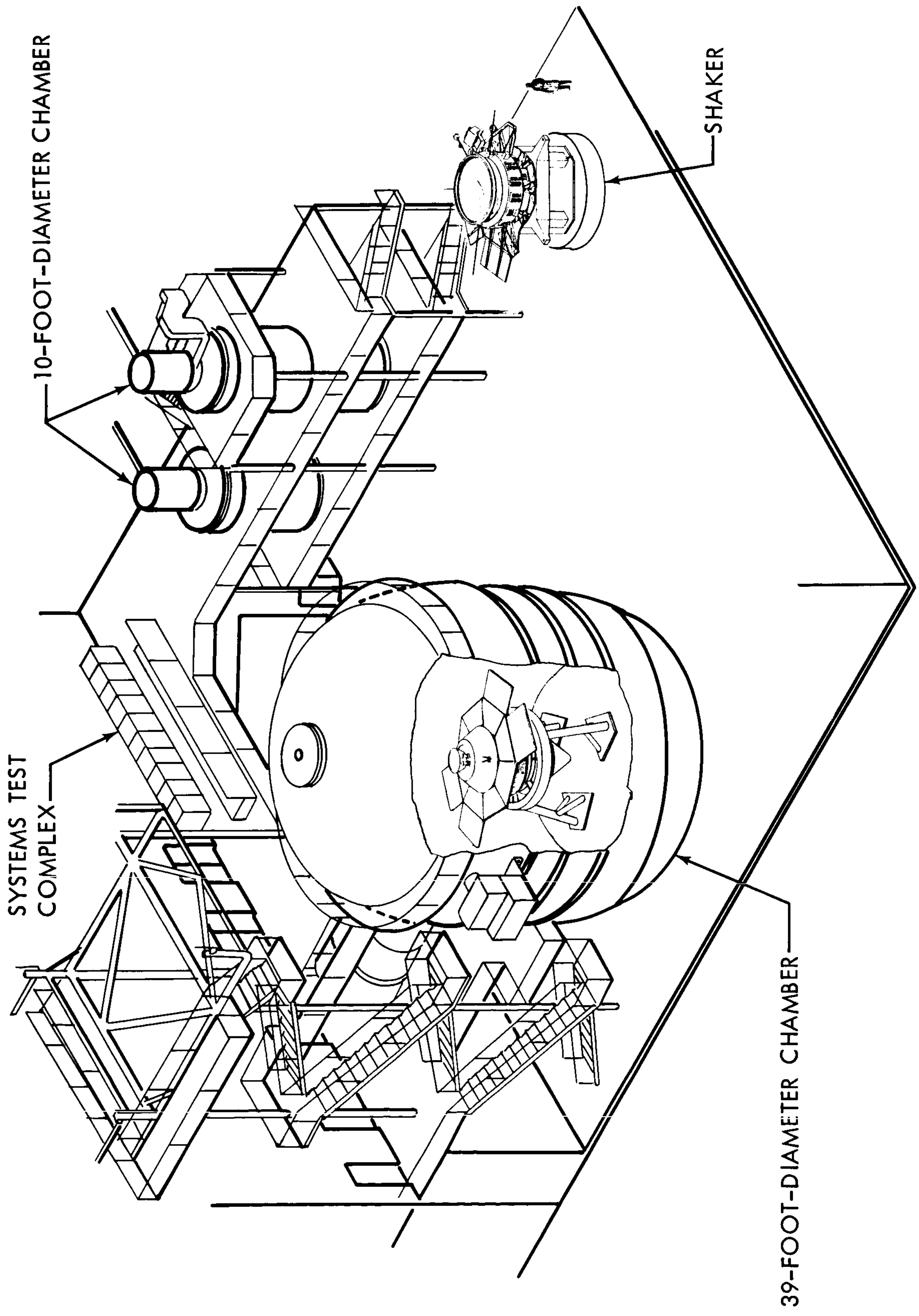


Figure 3.4-2: Kent Thermal - Vacuum, And Dynamic Test Facilities

The launch environment simulation facility contains a RISA (see Section 3.4.5.1); an 80- by 140-foot building housing the simulator chamber; overhead handling system of 20-ton capacity with 50-foot hook height; a spacecraft assembly stand; provision for routing interconnecting cabling from the STC; instrumentation and controls for directing the conditions of environment within the simulator chamber; 20- by 40-foot space for STC or LCE; and communications links to the STC at the assembly and test building.

The simulator chamber provides a vibration and acoustic test bed and has the following characteristics and capabilities: acoustic excitation equipment generating sound up to 160 db; evacuation equipment with pump-down capacity equivalent to that required to produce a simulated altitude of 100,000 feet; internal clear diameter of 30 feet, and 40-foot internal clear height; vibration excitation equipment; acoustic insulation to protect personnel outside the chamber; and telemetry link equipment for communication with the spacecraft in the chamber.

3.4.5.5 Propulsion Interaction Test Facility

This test facility is provided so that verifications may be made that:

1) The autopilot is capable of maintaining and controlling spacecraft attitude during firing of the midcourse propulsion system;
2) The dynamic properties of the spacecraft structure do not have a harmful effect on autopilot performance.

The chamber in this facility in which testing occurs can be evacuated to a pressure of 10^{-2} mmHg. Additional information on this subject is contained in Paragraph 3.3.6.6 of this volume.

3.4.5.6 Ancillary Test Facilities

The Boeing Company has existing test facilities at the Tulalip Test Site (north of Seattle), and at the Boardman Test Site (near Boardman, Oregon). Both have capabilities applicable to certain Voyager special test requirements:

1) Tulalip facilities will accommodate propulsion subsystem firing tests and spacecraft-capsule (or other) separation tests with minor facility changes.
2) By additions and modifications, orbit-insertion-motor testing can be accommodated at the Boardman site.
3) An antenna test range can be established on Boeing property at Tulalip, Boardman, or Seattle.

Cleanliness is not a requirement in these ancillary test areas.

3.4.5.7 STC and LCE Electrical Requirements

Main power for the STC and LCE will be supplied from the commercial power company serving the area in which the equipment is located.

The normal power source for the STC and LCE will be supplied by a two-winding transformer, which will provide the necessary isolation and voltage transformation. This will consist of a 3-phase, 4-wire, 60-cycle, 4160 - 208/120 volt, Delta-Wye connected dry-type transformer. This transformer will be equipped with two $\pm 2\frac{1}{2}$ percent high voltage taps and a fused primary load break switch. Power regulation requirements on the voltage and frequency are ± 5 and ± 1 percent, respectively.

An emergency main power source will be provided by an engine-generator set. The generator is a 208/120-volt, 3-phase, 60-cycle, machine designed to automatically pick up the load whenever the normal power source falls outside tolerances. The transfer from normal to emergency power does not exceed 0.5 second. Batteries supply interim power for critical items.

A distribution and control panel provides the functions of circuit protection, switching, and distribution of the main power. Also required for the LCE is a fail-safe power source to supplement the normal and emergency power, or during switching intervals, the spacecraft is placed in a safe condition and power for monitoring selected critical subsystems is provided.

The neutrals of the transformer and the generator will be tied to a common grounding grid. The cabinets of the STC and LCE equipment are tied to a separate ground system.

Table 3.4-3 lists the estimated power requirements at the various facilities.

TABLE 3.4-3 ESTIMATED POWER REQUIREMENTS (KVA)

Seattle	STC	LCE	OTHER
Assembly and Test Building	225	10	
Magnetic Test Facility	6	5	200
Space Simulation Facility	75	10	12
Launch Environment Simulation Facility	10	5	*
Propulsion Interaction Test Facility	10	10	*

*To be determined

3.4.6 Kennedy Space Center Facilities

The facilities baseline for the Kennedy Space Center is based on area requirements determined by the top-level flows. Cleanliness considerations exert major influence on the facility design. Where possible, dual usage of areas for tests are made to keep handling at a minimum. Three significant areas--the assembly and checkout area, the explosive safe area, and the pad area--have been evaluated.

Voyager processing at the KSC is accomplished in the following three areas.

1) The spacecraft checkout facility is located in the industrial area. The spacecraft is received and inspected, and subsystems are installed and checked by the STC.
2) The final assembly and checkout facility is located in an explosive safe area (ESA) where the propellants are loaded and the live orbit-injection motor and the pyrotechnics are installed. Final checkout, weight and balance, alignment, encapsulation, and decontamination are also accomplished. A magnetic test facility is in a separate building that complies with ESA requirements.
3) The encapsulated Planetary Vehicles are assembled to the launch vehicle at the pad facility in preparation for launch.

Table 3.4-4 lists the assembly and checkout functions performed at each facility. Figure 3.4-3 is a conceptual representation of facilities required at Cape Kennedy.

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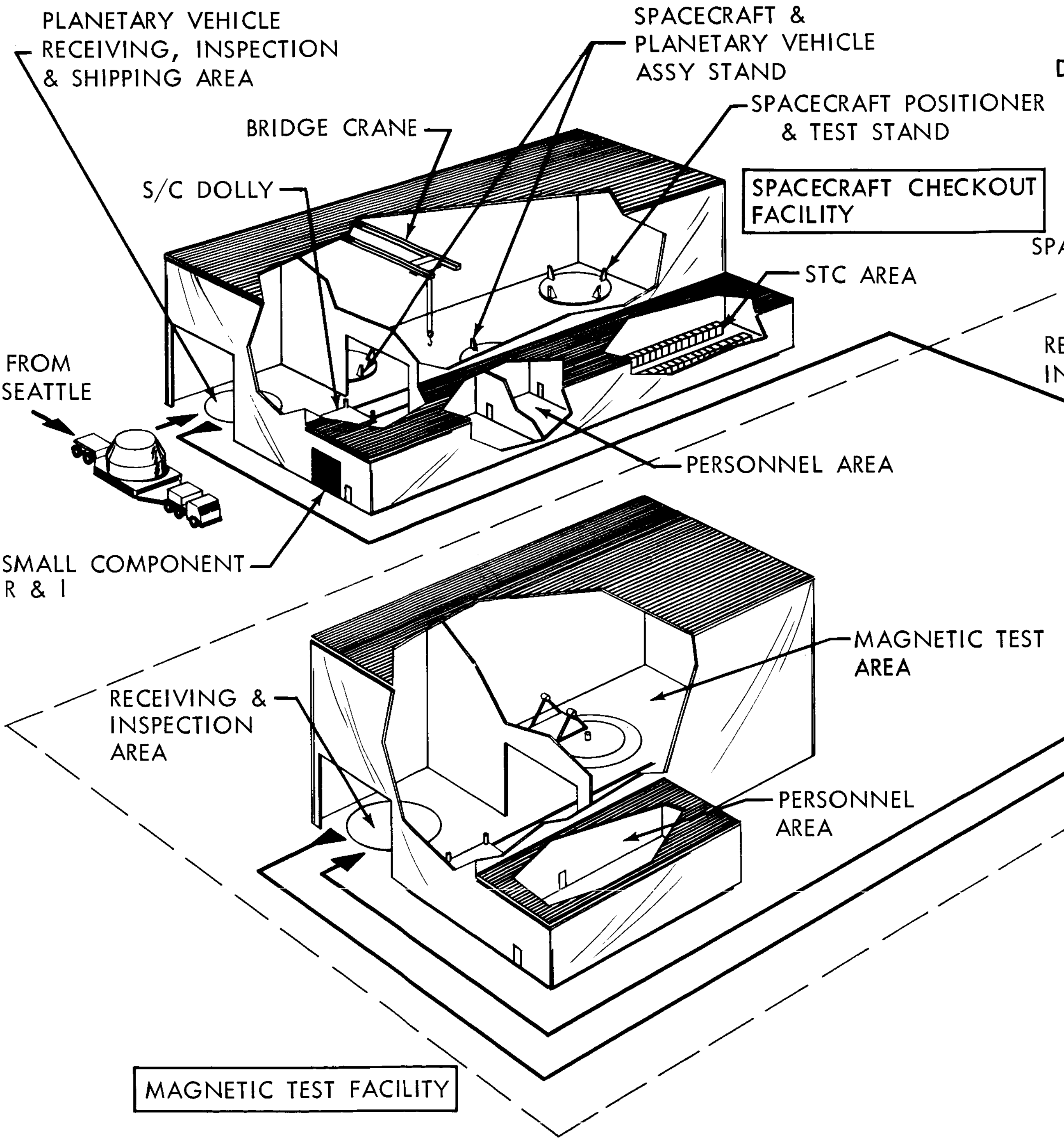
PLANETARY VEHICLE RECEIVING, INSPECTION & SHIPPING AREA
SPACECRAFT & PLANETARY VEHICLE ASSY STAND
SPACECRAFT POSITIONER & TEST STAND
BRIDGE CRANE
S/C DOLLY
SPACECRAFT CHECKOUT FACILITY
STC AREA
FROM SEATTLE
PERSONNEL AREA
SMALL COMPONENT R & I
MAGNETIC TEST AREA
RECEIVING & INSPECTION AREA
PERSONNEL AREA
MAGNETIC TEST FACILITY

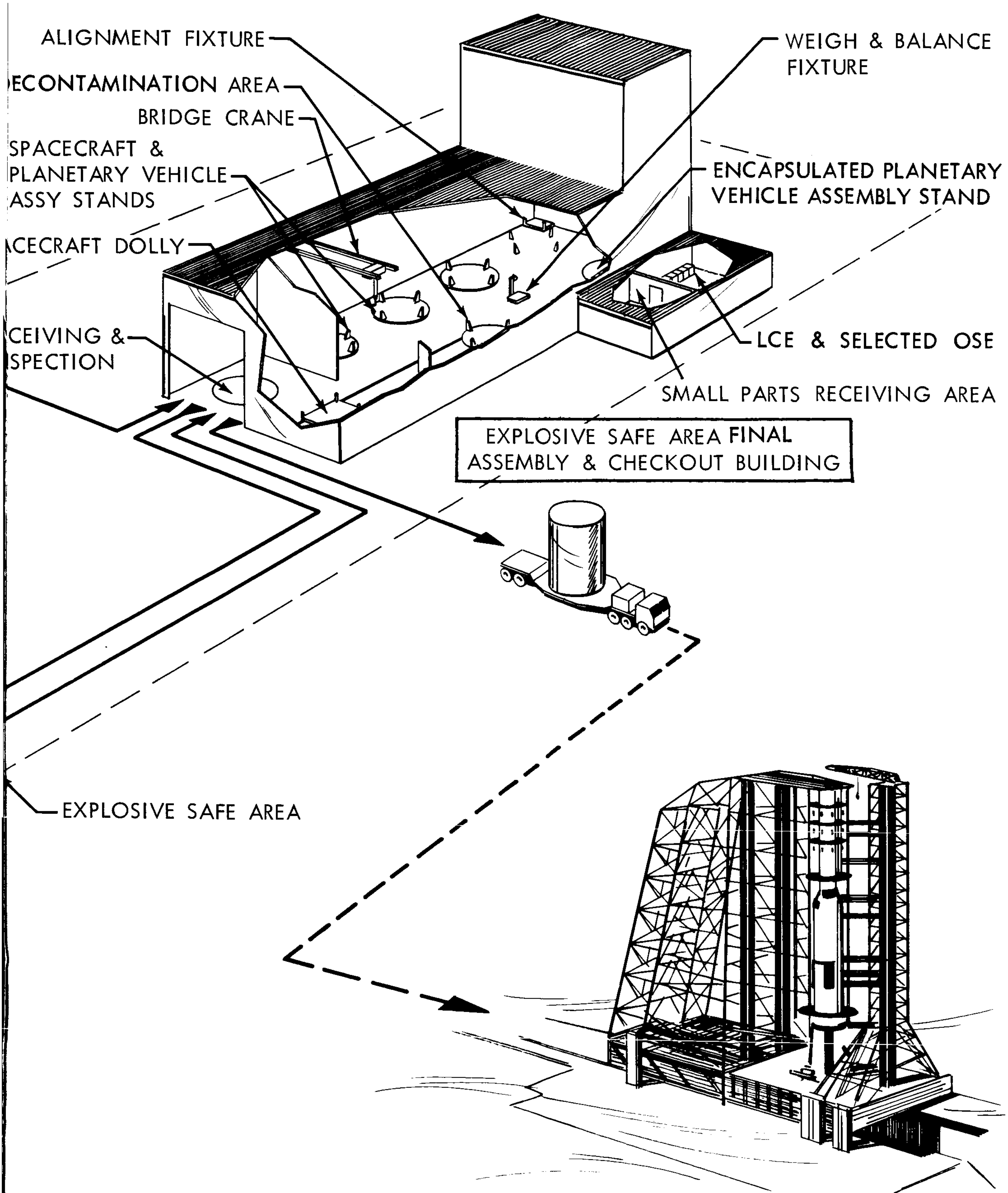

Figure 3.4-3: Kennedy Space Center

2

Table 3.4-4: KSC FACILITY OPERATION PROVISIONS

SPACECRAFT CHECKOUT FACILITY	EXPLOSIVE SAFE AREA	PAD FACILITY
Initial power application	Receiving & inspection alignment	LCE test
Subsystem tests	Internal decontamination of fluid systems in orbit insertion motor system and reaction-control propulsion system	Launch vehicle mating
Intersubsystem tests	Weight & balance	
Telemetry channel calibration	Spacecraft capsule mating	
System test	Planetary Vehicle-nose fairing mating	
Spacecraft-launch facility interface test	Propellant servicing	
Dummy run countdown	Load live orbit-injection engine	
Parameter variation	Spacecraft verification tests	
Assembly (i.e., solar panels, etc.)		
RF loop tests	SYSTEM MAGNETIC TEST FACILITY	
Final inspections	Perming	
Shipping preparations	Mapping	
	Deperming	
	(for S/C & PV)	

3.4.6.1 Spacecraft Checkout Facility

Receiving, Inspection, and Shipping Area (RISA)--The RISA is identical in performance and functions to the RISA in the central STC (Section 3.4.5.1) and has the same physical description.

Assembly and Checkout Area--The assembly and checkout area receives the spacecraft on the spacecraft dolly and transfers it to the assembly stand; provides for the handling of the spacecraft, capsule, dummy orbit-insertion engine, and various subassemblies; provides for assembly of spacecraft; supports testing by the STC; and provides overhead handling to move spacecraft from station to station.

The assembly and checkout area is illustrated in Figure 3.4-3 and requires approximately 8000 square feet of floor space. It is serviced by a bridge crane.

Construction meets Class-100,000 clean room requirements.

Other Areas--Support area provides space for offices, rest rooms, etc. Provisions are made for routing of interconnecting cabling for tests under the floor in the general test area, the encapsulated Planetary Vehicle test area, and the system test control area.

Also provided is a support area adjacent to the assembly and checkout area providing personnel "change" rooms; SSTE areas to support the system testing; storage for subassemblies, subsystems, and spares; and storage for OSE between periods of use. A system test control area is located adjacent to the spacecraft positioner and test stand. OSE, including STC, LCE, and MDE, is contained in this area. Test operations are commanded, controlled, and monitored from this area.

3.4.6.2 Explosive Safe Area

The ESA is separated from the spacecraft checkout facility of Section 3.4.6.1 to permit handling and checkout of the orbit-injection engine and servicing of the propulsion system. The requirements for an ESA as defined in AFETRM 127-1, "Range Safety Manual," apply to this area.

Receiving, Inspection, and Shipping Area (RISA)--The RISA described in Section 3.4.5.1 except it must also meet ESA requirements.

Final Assembly and Checkout Area--The functions performed in the final assembly and checkout area are in accordance with AFETRM 127-1. The

Planetary Vehicle is handled with a live orbit-injection motor, loaded and pressurized propulsion system, and pyrotechnics and arming devices. The area provides space for handling the spacecraft, capsule, Planetary Vehicle, and nose fairing; handling parts, subassemblies, or subsystems of the flight articles; final aligning Planetary Vehicle components; decontaminating the propulsion subsystem and reaction-control system; encapsulating the Planetary Vehicle; propellant servicing of the Planetary Vehicle; ordnance servicing of the Planetary Vehicle; installation and alignment of propulsion subsystem elements; weighing and balancing the Planetary Vehicle; and flight spacecraft systems verification.

The final assembly and checkout area requires 20,000 square feet of space.

The high-bay section of the final assembly and checkout area provides for Planetary Vehicle encapsulation, and for the mating test of the two Planetary Vehicles.

For safety reasons, this building is equipped with spark proof floors, adequate grounding connections, personnel emergency showers, emergency eyewash fountains, fire extinguishers, explosion-proof electrical fixtures, and a vapor detection system.

For magnetic mapping, the Planetary Vehicle is installed in its shipping container and moved to the magnetic test facility. After mapping is complete, the Planetary Vehicle is returned to the final assembly and checkout building for final checkout and encapsulation. Section 3.4.5 describes the magnetic test equipment and facility requirements. The KSC magnetic-mapping facility meets ESA requirements as defined by AFETRM 127-1.

3.4.6.3 Pad Facilities

The Voyager program uses the launch pad facilities of Launch Complex 39. Trade studies were performed on alternate assembly sequences; these determined that the encapsulated Planetary Vehicles are individually installed on the launch vehicle at the launch pad. The facilities support the following functions:

1) Hoisting planetary vehicles to booster interface;
2) Mechanical mating;
3) Spacecraft systems tests;
4) Combined system tests;
5) Launch preparation;
6) Countdown and launch.

The general arrangement of the pad facilities is shown in Figure 3.4-3.

Access roads to the spacecraft pickup area are required and must permit use of a transport vehicle with an overall length of 75 feet and an overall width of 24 feet. The spacecraft segment pickup area is located within 85 feet of the centerline of the launcher tower to permit use of the existing crane.

The launch tower crane lifts the spacecraft segments from the transport vehicle onto the launch vehicle or previously assembled Planetary Vehicle segment. This requires a crane capacity of 32,500 pounds at a reach of 85 feet and a vertical hook travel of approximately 420 feet.

The assembly methods trade study established the feasibility of modifying the crane to these performance requirements. Modifications consist generally of strengthening the crane boom (truss) structure and increasing the weight of the crane counterweight.

Three access platforms are required to permit access to the spacecraft mating surfaces. They should be located: (1) at the encapsulated Planetary Vehicle/launch vehicle interface; (2) at the interface between the two planetary vehicles; and (3) at the interface between Planetary Vehicle 2 and the nose fairing.

A service arm is required to carry ground power and control/monitor lines from the mobile launch tower to the Planetary Vehicles. Provisions are made for routing the cables from the equipment rooms in the base of the tower to the service arm.

3.4.6.4 STC and LCE Electrical Requirements

The electrical requirements for the STC and LCE are described in Section 3.4.5.7. Estimated power requirements for KSC are given in Table 3.4-5.

Table 3.4-5: ESTIMATED POWER REQUIREMENTS (KVA)

Kennedy Space Center	STC	LCE	Other
Spacecraft Checkout Facility	150	10	10
Explosive Safe Area	15	10	10
Magnetic Test Facility	6	5	120
Launch Pad	6	10	7
LCC		10	

3.4.7 Deep Space Network (DSN) Facilities

The facilities requirements for the DSN provides for MDE to be installed in the DSIF and SFOF. The location of the facilities are: SFOF, Pasadena, California; DSIF 71, Cape Kennedy, Florida; DSIF 72, Ascension Island; DSIF 61, Madrid, Spain; DSIF 42, Tidbinbilla, Australia; DSIF 13, Goldstone, California.

3.4.8 Goldstone Facilities

Facilities are required at Goldstone to facilitate testing the Planetary Vehicle communication system.

Requirements include a receiving, inspection, and shipping area, and a shielded test room. A crane is required for moving the Planetary Vehicle and placing it on the test stand.

3.5 SPACECRAFT MISSION-DEPENDENT EQUIPMENT

The spacecraft Mission-Dependent Equipment (MDE) consists of all the unique hardware and software items required to accommodate the Voyager mission. The unique equipment is located at the Deep Space Instrumentation Facility (DSIF) and the Space Flight Operations Facility (SFOF). The MDE will be integrated with the 1971 configuration of the Deep Space Network (DSN) to provide overall support of the Voyager missions.

3.5.1 Summary

The required MDE functions are as follows:

1) Buffer and process the telemetry data stream into the data processing systems at the DSIF and SFOF:
    a) Demodulate the separate data subcarriers and prepare them for further processing,
    b) Buffer and reformat serial pulse-code-modulated (PCM) data for entry into the telemetry and command data subsystem (TCD) for data transfer to the SFOF,
    c) Process the block-encoded data for shipment via tapes or transmission via ground communication lines to the SFOF for decoding and data reconstruction.
2) Support preflight checkout and mission simulation.

Mission-dependent software as described in Section 3.6 is required for processing and control of:

1) Telemetry and command data (TCD);
2) Mission integration and control (MIC);
3) Spacecraft performance analysis and command (SPAC);
4) Flight-path analysis and command (FPAC).

The MDE required at a typical DSIF site and the SFOF is shown in Figure 3.5-1. The requirement to simultaneously accommodate two Planetary Vehicles within a DSIF site is provided by two identical sets of MDE. The mission-dependent software for the TCD subsystem is described in Section 3.6. With the MDE and the mission independent equipment (MIE), it will provide the operational capability to accommodate the Voyager missions.

Each set of MDE at a DSIF site demodulates the lower frequency telemetry channel in each of the four telemetry operating modes. The lower channel contains the spacecraft-engineering, capsule-engineering, and cruise-science data. The recovered PCM data train is buffered for entry into the TCD subsystem, where it is selectively processed for real-time presentation and for transmission to SFOF. The MDE also provides for demodulation of the upper-frequency telemetry channel, which contains the planetary-science data. The output of the upper subcarrier demodulator is coverted to a parallel digital format in the block-encoded-data processor and recorded on magnetic tape for transport to SFOF for decoding and data processing.

The DSIF MIE acquires predetection recording at a 10-Mc center frequency. Also, a postdetection recording of the lower-frequency subcarrier baseband is acquired to avoid loss of data in the event of a failure in the postdetection operations. The predetection recorder (Figure 3.5-1) is used to ensure acquisition of planetary-science data. It is not used during the interplanetary cruise phase.

The MDE at the SFOF site provides the block-decoding capability necessary to recover the science data contained on the upper telemetry channel. The DSIF 71 MDE includes the SFOF MDE for prelaunch checkout and simulation.

No requirement for near-real-time reduction of planetary-science data is currently identified. If such a requirement is identified in Phase IB, each DSIF station will be implemented with a block decoder so that scientific data may be transmitted to SFOF over high-speed data links.

3.5.2 Applicable Documents

The applicable Operational Support Equipment (OSE) and MDE documents are listed in Section 2.1. Additional references for MDE are:

1) EPD 23, "SFOF Data Processing System," July 20, 1964;
2) "Space Programs Summary No. 37-24, Volume III," November 30, 1963.

3.5.3 Design Constraints and Requirements

The design constraints and requirements for the Voyager Spacecraft System MDE are those imposed by the DSN, the spacecraft telecommunications subsystem, and the Mission Operations System (MOS).

3.5.3.1 Constraints

The constraint imposed on the MDE by the foregoing Voyager program elements are as follows.

1) The MDE will be compatible with the functional constraints imposed by the spacecraft telemetry system.
2) The functional, electrical, and physical interfaces of the MDE will mate with existing or planned MIE at the DSIF and SFOF.

3) The location of the MDE will be compatible with existing or planned facilities and MIE as required.

4) The MDE will be compatible with the functional constraints imposed by the MOS.

3.5.3.2 Requirements

The requirements for the MDE are as follows.

1) The DSIF MDE will accept the postdetection telemetry signals from the DSIF station receiver and process them for functional and electrical compatibility with the TCD subsystem and the data processing and recording subsystem. The DSIF MDE will provide status monitoring and control in conjunction with the digital instrumentation and station monitor and control subsystem.

2) The DSIF MDE will accept the predetection telemetry signals from the DSIF station receiver and process them for functional and electrical compatibility with the TCD subsystem and the data processing and recording subsystem. The processing of planetary-science data may be limited to that necessary for checkout and simulation.

3) The SFOF will accept the DSIF postdetection tape recordings and convert them to be functionally and electrically compatible with the telemetry processing station for follow-on processing.

3.5.4 Trade Studies

The presently preferred design for MDE provides biorthogonal block-decoding capability only at the SFOF and DSIF 71. An additional study will be conducted in Phase 1B to consider implementing a biorthogonal block decoder (to recover the planetary-science data stream on the upper-frequency telemetry channel) at each DSIF site versus implementing decoders only at SFOF and at DSIF 71. The parameters are as follows:

1) The SDS 920 computer, because of speed limitations, cannot perform all decoding at the DSIF sites, although limited decoding for check-out purposes can be accomplished.
2) Provision of a biorthogonal block decoder at SFOF, only to perform the decoding function, precludes reduction of planetary-science data until magnetic tapes are transported from the DSIF sites to the SFOF. Hence, reaction time is long for data received at any station except Goldstone.
3) Provision of a biorthogonal block decoder at each DSIF site to perform the decoding function permits transmission of the planetary-science data to the SFOF over high-speed data links, thereby providing a shorter reaction time, but requiring additional equipment.

Final resolution of the study is contingent on further definition of the Voyager mission as directed by JPL. Presently, the preferred design provides operational block-decoding capability only at the SFOF and DSIF 71.

3.5.5 Selected Configuration and Functional Descriptions

The selected configuration of the Voyager MDE is shown in Figure 3.5-1. This design satisfies the requirements imposed by the primary objective of the Voyager mission to acquire and recover cruise and planetary-science data.

Subcarrier telemetry data, derived from the DSIF receiver, enter the MDE and are routed to the correct subcarrier demodulators by the telemetry-

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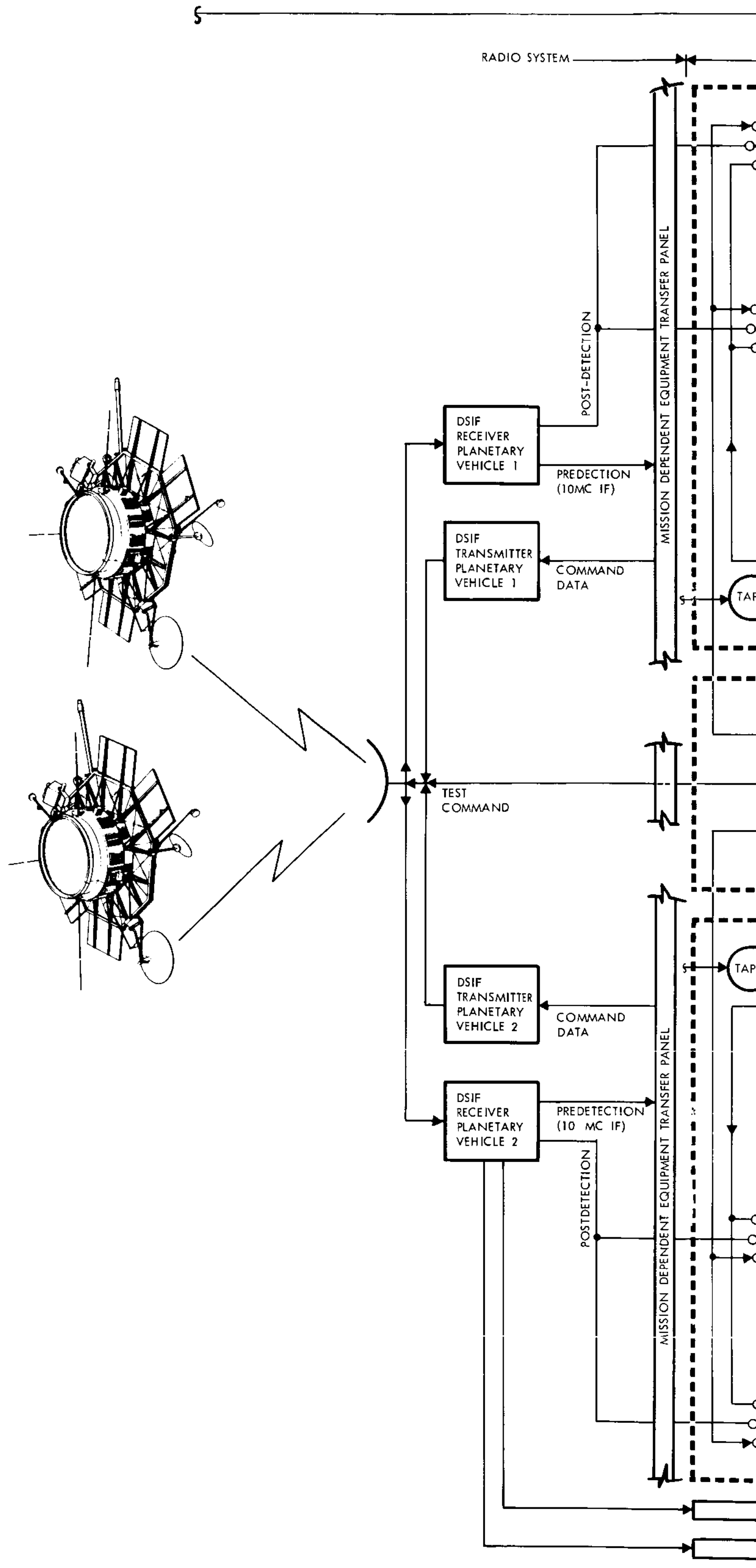
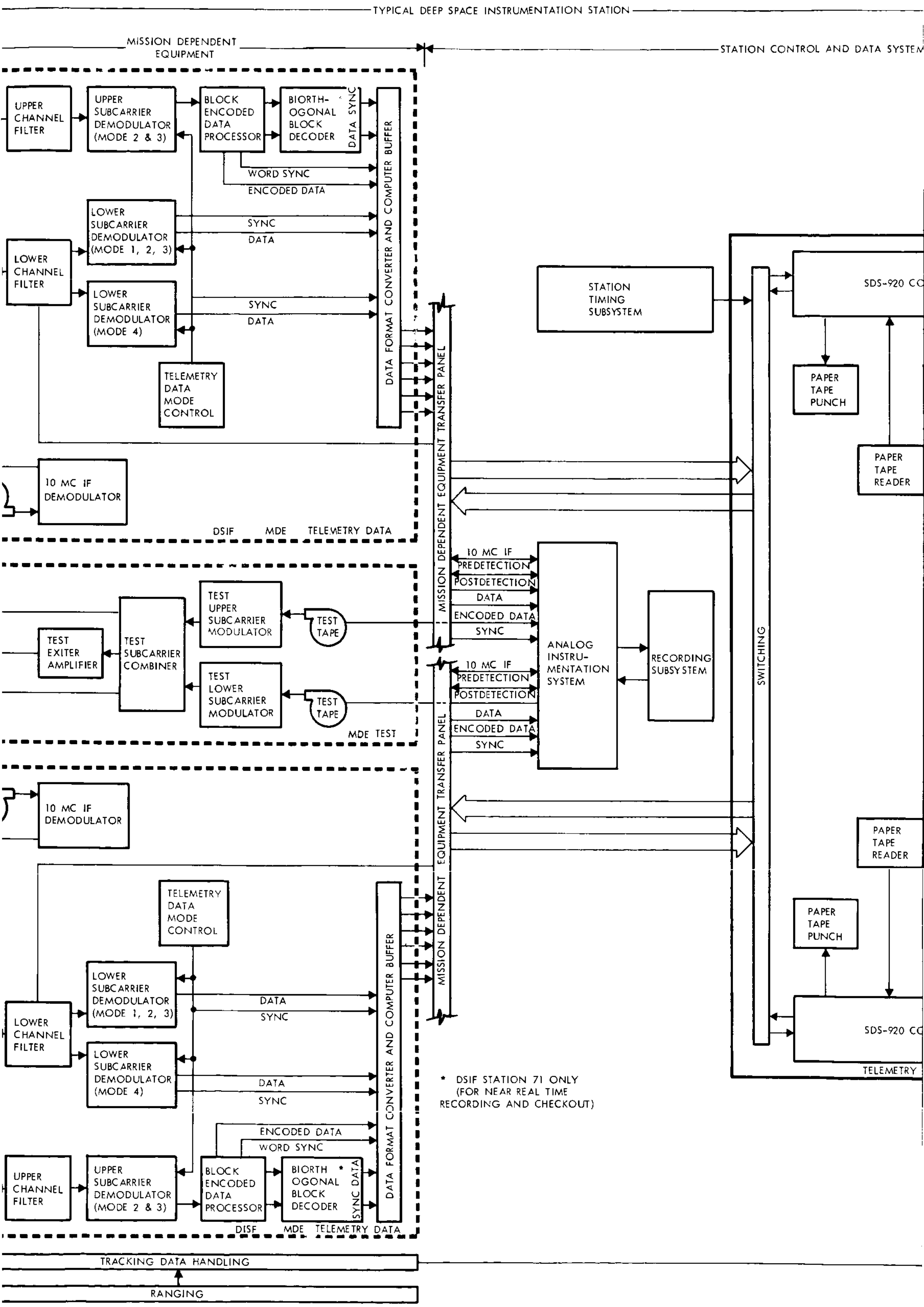
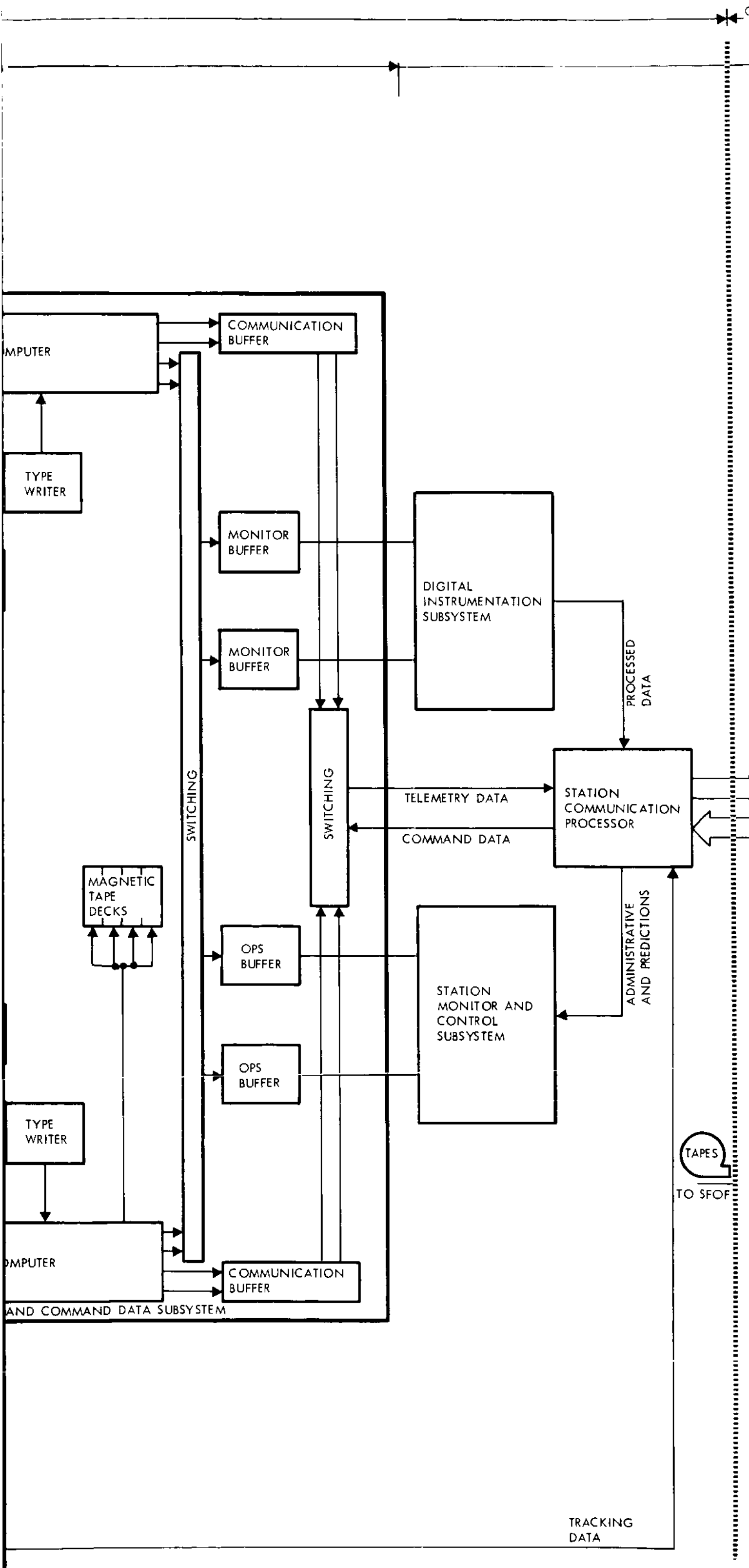
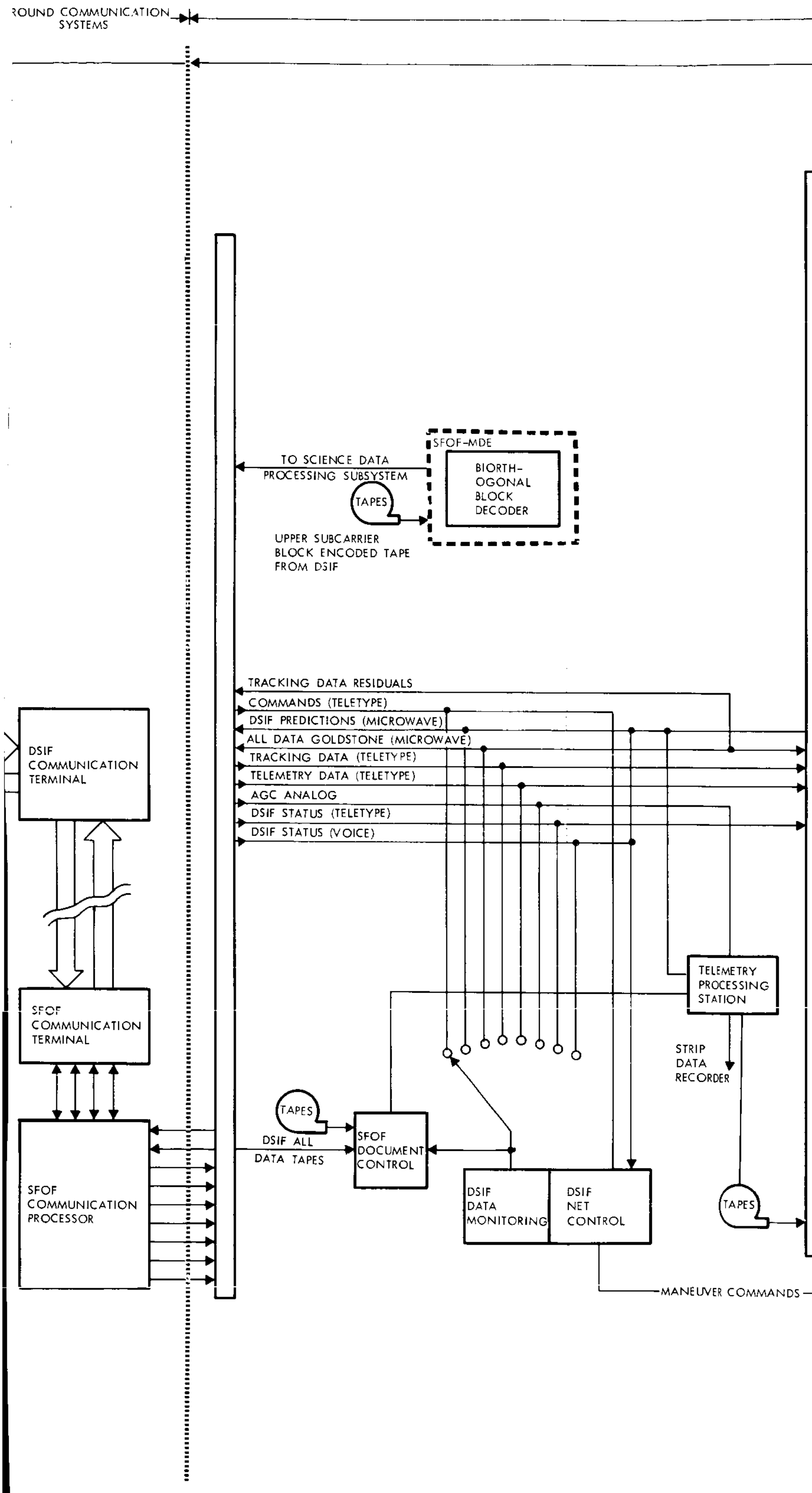

RADIO SYSTEM
MISSION DEPENDENT EQUIPMENT TRANSFER PANEL
POST-DETECTION
DSIF RECEIVER PLANETARY VEHICLE 1
PREDECTION (10MC IF)
DSIF TRANSMITTER PLANETARY VEHICLE 1
COMMAND DATA
TEST COMMAND
DSIF TRANSMITTER PLANETARY VEHICLE 2
COMMAND DATA
DSIF RECEIVER PLANETARY VEHICLE 2
PREDETECTION (10 MC IF)
POSTDETECTION
MISSION DEPENDENT EQUIPMENT TRANSFER PANEL



TYPICAL DEEP SPACE INSTRUMENTATION STATION
MISSION DEPENDENT EQUIPMENT
STATION CONTROL AND DATA SYSTEM
UPPER CHANNEL FILTER
UPPER SUBCARRIER DEMODULATOR (MODE 2 & 3)
BLOCK ENCODED DATA PROCESSOR
BIORTH-OGONAL BLOCK DECODER *
DATA SYNC
WORD SYNC
ENCODED DATA
LOWER SUBCARRIER DEMODULATOR (MODE 1, 2, 3)
SYNC
DATA
LOWER CHANNEL FILTER
LOWER SUBCARRIER DEMODULATOR (MODE 4)
TELEMETRY DATA MODE CONTROL
DATA FORMAT CONVERTER AND COMPUTER BUFFER
MISSION DEPENDENT EQUIPMENT TRANSFER PANEL
10 MC IF DEMODULATOR
DSIF MDE TELEMETRY DATA
STATION TIMING SUBSYSTEM
SDS-920 CO
PAPER TAPE PUNCH
PAPER TAPE READER
TEST UPPER SUBCARRIER MODULATOR
TEST TAPE
TEST EXITER AMPLIFIER
TEST SUBCARRIER COMBINER
TEST LOWER SUBCARRIER MODULATOR
MDE TEST
10 MC IF
PREDETECTION
POSTDETECTION
DATA
ENCODED DATA
SYNC
ANALOG INSTRU-MENTATION SYSTEM
RECORDING SUBSYSTEM
SWITCHING
TELEMETRY
* DSIF STATION 71 ONLY (FOR NEAR REAL TIME RECORDING AND CHECKOUT)
TRACKING DATA HANDLING
RANGING


MPUTER
COMMUNICATION BUFFER
TYPE WRITER
MONITOR BUFFER
MONITOR BUFFER
DIGITAL INSTRUMENTATION SUBSYSTEM
PROCESSED DATA
SWITCHING
SWITCHING
TELEMETRY DATA
COMMAND DATA
STATION COMMUNICATION PROCESSOR
MAGNETIC TAPE DECKS
OPS BUFFER
STATION MONITOR AND CONTROL SUBSYSTEM
ADMINISTRATIVE AND PREDICTIONS
OPS BUFFER
TYPE WRITER
TAPES
TO SFOF
MPUTER
COMMUNICATION BUFFER
AND COMMAND DATA SUBSYSTEM
TRACKING DATA

3



ROUND COMMUNICATION SYSTEMS
SFOF-MDE
BIORTH-OGONAL BLOCK DECODER
TO SCIENCE DATA PROCESSING SUBSYSTEM
TAPES
UPPER SUBCARRIER BLOCK ENCODED TAPE FROM DSIF
TRACKING DATA RESIDUALS
COMMANDS (TELETYPE)
DSIF PREDICTIONS (MICROWAVE)
ALL DATA GOLDSTONE (MICROWAVE)
TRACKING DATA (TELETYPE)
TELEMETRY DATA (TELETYPE)
AGC ANALOG
DSIF STATUS (TELETYPE)
DSIF STATUS (VOICE)
DSIF COMMUNICATION TERMINAL
SFOF COMMUNICATION TERMINAL
TELEMETRY PROCESSING STATION
STRIP DATA RECORDER
TAPES
DSIF ALL DATA TAPES
SFOF DOCUMENT CONTROL
SFOF COMMUNICATION PROCESSOR
DSIF DATA MONITORING
DSIF NET CONTROL
TAPES
MANEUVER COMMANDS

4

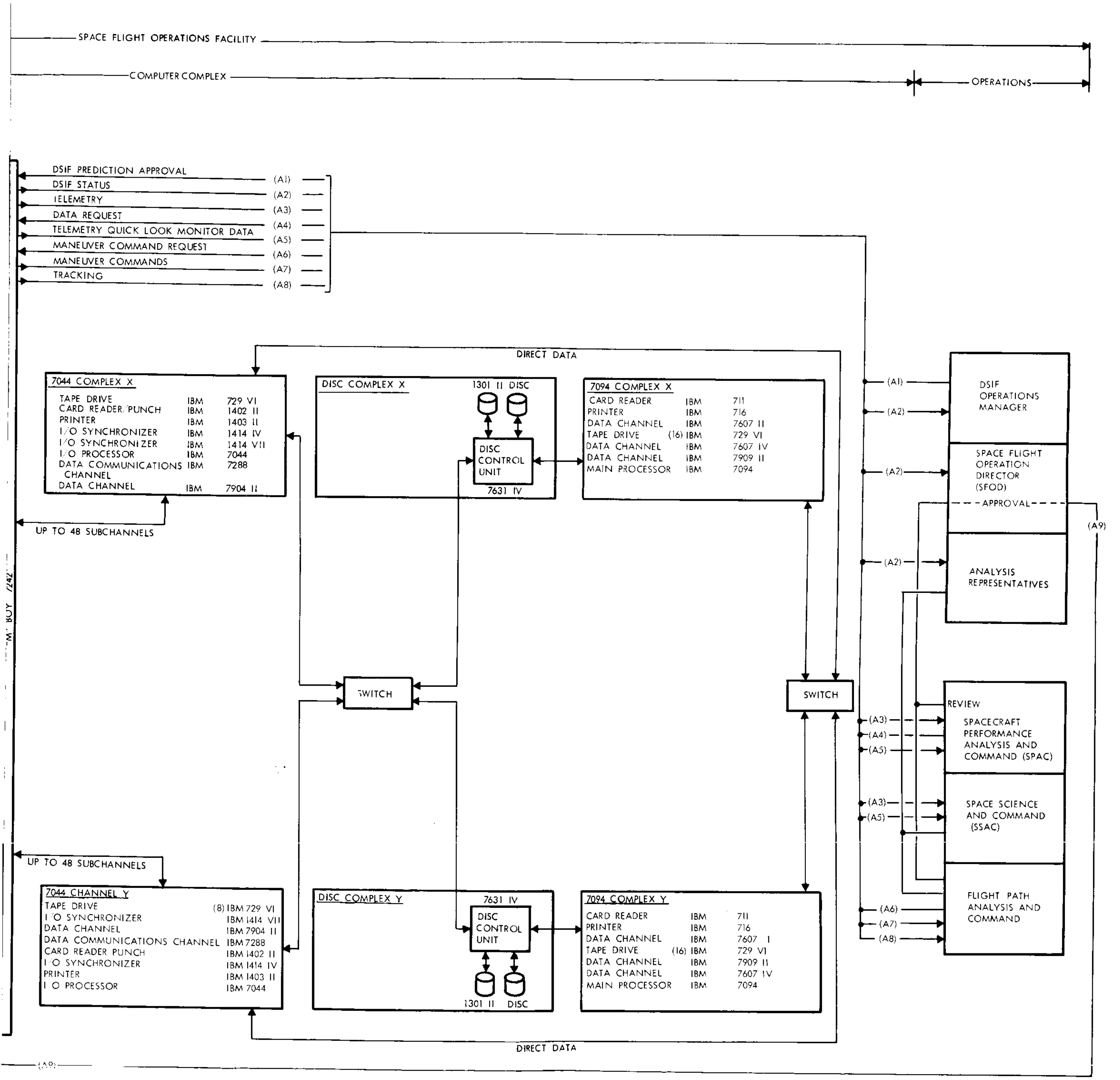


Figure 3.5-1: Integrated DSN-MDE Configuration

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data mode control. The outputs of the lower subcarrier demodulators pass through the MDE transfer panel to the DSIF TCD subsystem via the data format converter and computer buffer. The TCD subsystem processes the telemetry data for transmission to the SFOF via the communications processor and the ground communications system. At the SFOF, it is simultaneously recorded by magnetic-tape units and routed to the telemetry-processing station. The output from the upper subcarrier demodulator is recorded directly on magnetic tape at the DSIF for transport to the SFOF, where it is decoded and processed for analysis. At the DSIF site, the output of the upper subcarrier demodulator is also presented to the TCD so that a limited amount of biorthogonal block-encoded data can be processed to facilitate station checkout.

Both predetection and postdetection baseband recordings are made at the DSIF as protection against loss of data. These tapes are transported to the SFOF for playback through the Goldstone facility and subsequent entry to the SFOF through normal ground communications system channels.

A functional description of each MDE item shown in Figure 3.5-1 is contained in the following paragraphs.

Lower Subcarrier Demodulator, Modes 1, 2, and 3--The lower subcarrier demodulator (single channel) for Telemetry Modes 1, 2, and 3 recovers the serial PCM data train and derives a bit-rate clock from the biphase-modulated telemetry subcarriers. The input signals are received from the DSIF receiver. In the case of Goldstone, they are also received from the reproduction of predetection or lower-channel baseband recordings.

Lower Subcarrier Demodulator, Mode 4--The subcarrier demodulator (dual channel) for Telemetry Mode 4 recovers the serial PCM data train and derives a bit-rate clock from the coherent dual-channel telemetry subcarriers. The input signal consists of two subcarriers; the lower is a pseudo-noise-coded synchronization signal, and the upper is a biphase-modulated data signal. The signals are received from the DSIF receiver. Again, in the case of Goldstone, they are derived from reproductions of predetection or lower-channel baseband recordings.

The demodulators provide frame synchronization and generate master timing formats for all spacecraft telemetry modes. They furnish the data format converter and computer buffer with word-rate and frame-rate timing signals to decommutate telemetry data.

The data format converter and computer buffer converts the telemetry data words into a format suitable for entry into the DSIF TCD subsystem and provides the interface buffer.

Upper Subcarrier Demodulator, Modes 2 and 3--The upper subcarrier demodulator (Modes 2 and 3) removes the subcarrier modulation and provides that an 11-bit word (10 bits plus sign) is generated for each received PCM bit. The 11 bits describe the analog output of an integrating-type bit detector that is reset at the beginning of each bit. Word rate for the coded data on the upper subcarrier is derived from the lower subcarrier demodulator. The outputs from the upper subcarrier demodulator are recorded on magnetic tape for transport to SFOF. The input signal is derived from the DSIF receiver and, in the case of Goldstone, also from the reproduction of the predetection recording.

The auxiliary test equipment provides the capability of checking out the DSIF station MDE.

3.5.6 Interfaces

The major interfaces for the MDE are at the DSIF and SFOF (see Section 2.1). At the DSIF, the MDE must interface with the following items, as shown in Figure 3.5-1.

1) DSIF S-band receiving equipment provides interface telemetry signals for the MDE.
2) DSIF data processing and recording subsystem will be used to obtain predetection recording of all planetary-science telemetry data.
3) DSIF digital instrumentation and station monitor and control subsystem will provide for status monitoring and control.
4) DSIF TCD subsystem accepts inputs from the MDE.

The MDE at the SFOF must interface with the patch panel located at the telemetry processing station, which is a part of the SFOF (Figure 3.5-1).

3.5.7 Performance Parameters

Mode 1 telemetry data is recovered by the MDE at each DSIF site and transferred to the TCD subsystem. Mode 1 employs phase-shift keying (PSK) modulation for the conveyance of telemetry information. The channel consists of an 80-(bps)-nonreturn to zero (NRZ) PCM data train biphase ($\pm\pi/2$ radians) modulated on a 1440-cps subcarrier. The characteristics of the data train are:

1) Word length--7 bits.
2) Master frame length--26 words.

3) Master frame sync--the first word of each master frame is a 7-bit Barker word.

4) Coding--NRZ such that one phase (level) represents a binary "1" and the other phase (level) represents a binary "0".

The outputs from the demodulator consist of the reconstructed data train, the data bit rate, the data word rate, and the data frame rate. The reconstructed data train has all ambiguity removed except that permitted by the allowable bit errors. The bit rate, word rate, and frame rate are synchronized to the beginning of the bit, word, and frame, respectively, as required for entry of the data train into the data format converter and computer buffer.

The demodulator detects and reconstructs the data train with an error rate of no more than 5 per 1000 bits when the ratio of signal power to Gaussian noise power per cycle of bandwidth at the input to the detector is 25.7 db or greater.

Mode 2 modulation is partially recovered at the DSIF sites. Mode 2 uses a dual-channel modulation technique for the conveyance of telemetry data. The lower-frequency channel consists of a 288-bps NRZ PCM data train biphase modulated on a 1440-cps subcarrier. The data characteristics are the same as those of Mode 1.

The higher-frequency channel consists of NRZ PCM biorthogonal (16,5) block-encoded data biphase modulated on a 92,160-cps subcarrier. The characteristics of the data are:

1) Data word length--5 bits;

2) Code word length--16 bits;

3) Data bit rate--7,200 bps;

4) Code bit rate--23,040 bps;

5) Coding--Biorthogonal block (16,5) encoding of the data with an NRZ output such that one phase of the subcarrier represents a binary "1" and the other phase represents a binary "0."

6) Sync--One code word per cycle of the lower-frequency subcarrier.

Biorthogonal coding increases the transmission efficiency of PCM telemetry by coding the PCM planetary-science pulse train into a new pulse train less susceptible to contamination by noise. In a 16,5 system, the input PCM data train is broken into 5-bit data "blocks." Each block, which represents one of 32 (i.e., 2^5) possible combinations of binary 1's and 0's, is identified uniquely with one of 32 biorthogonal 16-bit code words. The orthogonal word is then transmitted in place of the 5-bit data block. The transmitted bit rate is 16/5 times the input PCM bit rate to preserve the data transmission rate. On the ground, the received signal is demodulated, and each of the possible 16-bit code words that could have been transmitted is multiplied by the noise-corrupted received signal. The most probable transmitted 16-bit code word is that which has the greatest product when multiplied by the received wave. Since each bit error in the 16-bit word changes the expected correlation voltage by 1/16 unit, several errors can be accommodated without having the data destroyed. Because the error tolerance is greater than the errors introduced by widening the bandwidth (by the ratio 16/5), a net gain in transmission efficiency is realized. Figure 3.5-2 illustrates the expected signal-to-noise gain from 16,5 biorthogonal coding. This figure is based on work at JPL as reported in pages 81-86 of the document listed as Item 2 in Section 3.5.2.

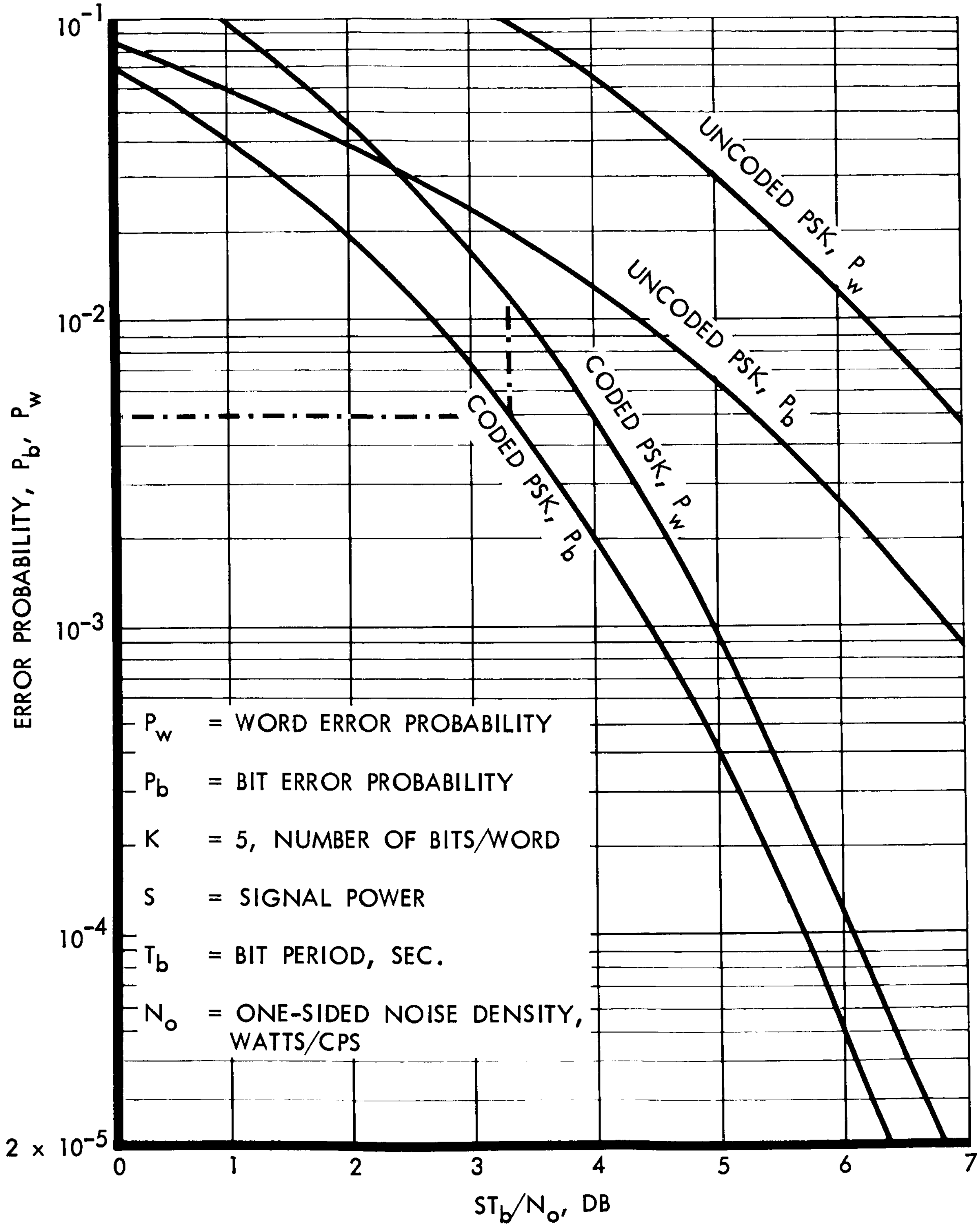


Figure 3.5-2: Bit And Word Error Probability For A (16,5) Biorthogonal Code

Two sets of outputs are provided by the demodulators at the DSIF. The reconstructed data train, data bit rate, data word rate, and data frame rate are derived from the demodulation of the lower-frequency subcarrier as in Mode 1 except that the error rate will be no more than 5 per 1000 bits when the ratio of signal power to noise power per cycle of bandwidth at the input of the demodulator is 31.3 db or more.

The second set of outputs are from the upper-frequency subcarrier. The data train is extracted by the subcarrier demodulators and a digital output representing the amplitude of each bit is provided. Each bit is detected by an integrator that operates during each bit period. The accumulation of the integrator is sampled and digitized at the end of each bit period. Digitization is to 11 bits (10 bits plus sign). The binary bit stream resulting from the digitization process (a stream at 11 times the received bit rate) is simultaneously routed to a tape recorder and to the data format converter and computer buffer.

The MDE at SFOF completes the recovery of planetary-science data from the upper subcarrier. The quantification of the demodulator output (a DSIF MDE item) is reproduced into the biorthogonal block decoder, which has an arithmetic unit for each of the 16-bit code words in the telemetry vocabulary. The reproduced data is simultaneously multiplied, on a bit-for-bit basis, by each of the 16-bit words, and the results are stored in accumulators until the operation on a word has been completed. The accumulators are then tested, and the one accumulator that shows the best correlation is selected as representing the 16-bit biorthogonal word that was most probably transmitted. The five data bits associated with this identified code word are then generated. The output of this unit is the planetary-science data stream, or its complement.

The error rate of this reconstructed data stream will be no more than 5 per 1000 bits when the ratio of signal power to noise power per cycle of bandwidth at the input to the demodulator (the DSIF MDE) is 43.3 db or more.

Mode 3 modulation is partially recovered at the DSIF sites. Mode 3 is identical to Mode 2, with the following exceptions.

The lower-frequency channel consists of a 60-bps NRZ PCM data train biphase modulated on a 240-cps subcarrier. The error rate of the reconstructed data will be no more than 5 errors per 1000 bits when the ratio of signal power to noise power per cycle of bandwidth at the input to the demodulator is 24.5 db or more.

The upper-frequency channel consists of a NRZ PCM biorthogonal planetary-science data train biphase modulated on a 92,160-cps subcarrier. The data bit rate is 1200 bps and the code bit rate is 3840 bps. The reconstructed data will have an error rate of no more than 5 per 1000 bits when the ratio of signal power to noise power per cycle of bandwidth at the input to the demodulator is 35.6 db or more.

Mode 4 data is recovered at the DSIF sites. Mode 4 uses a two-channel coherent-modulation technique for the transmission of telemetry data. The data channel is obtained by biphase modulating a 1.64-bps NRZ PCM data train on a 240-cps subcarrier. The characteristics of the data train are the same as for Mode 1. The sync channel is obtained by Modulo-2 adding a 120-bps subcarrier and a cyclic 511-bit pseudo-noise code generated at 120-bps. The data subcarrier, the sync subcarrier, and the pseudo-noise code are phase coherent.

The outputs from the demodulator consist of the reconstructed data train, the data bit rate, the data word rate, and the data frame rate. The reconstructed data train has all ambiguity removed within the limit of the bit error rate. The bit rate, word rate, and frame rate are to be synchronized to the beginning of the bit word and frame, respectively, as required for entry of the data train into the data format converter and computer buffer.

The demodulator detects and reconstructs the data train with an error rate of no more than 5 per 1000 bits when the ratio of signal power to noise power per cycle of bandwidth at the input to the detector is:

1) Sync channel--8.2 db or greater;
2) Data channel--7.7 db or greater.

3.6 COMPUTER SOFTWARE

The computer software described in this report includes the computer programs necessary for:

1) Mission-dependent functions of the Mission Operations System (MOS);
2) Computational support of spacecraft launch operations;
3) Operation of System Test Complex (STC) and subsystem Operational Support Equipment (SS OSE);
4) Spacecraft system test and subsystem test data reduction and analysis;
5) Computational support of trend analysis.

3.6.1 Summary

This section presents the objectives, requirements, constraints, and functional descriptions of the OSE software to support the Voyager 1971 program. The reliability, safety, and testing of that software are discussed. A fundamental objective is to provide adequate analytical capability for all levels of testing. Emphasis throughout the design effort has been given to developing maximum support to attainment of mission success and use of Mariner program experience. Another objective has been achievement of upward and downward compatibility and modularity so that processed data from any test can be precisely correlated with related test data from a test of both higher and lower levels of spacecraft hardware assembly.

The modular design standards assigned to these computer programs have also been selected to minimize redundant reprogramming of functions while at the same time contributing to the confidence that can be placed in these programs as they are used to support the successive activities of the Voyager program.

Computer program functional requirements have been analyzed in terms of the competing characteristics. Priorities have been placed on software competing characteristics, as follows:

1) Assurance of correct operation in all operating modes with all permutations of input data;
2) Probability of detecting submarginal performance or failure of Voyager spacecraft hardware;
3) Minimum reprogramming of identical and similar functions to support the various levels and sequences of tests and operations;
4) Flexibility of operation as required to facilitate spacecraft troubleshooting and special problem analysis;
5) Ease of change to accommodate revisions in test procedures and spacecraft design improvement or growth.

The above-described computer software characteristics are based on JPL's documented Mariner experience. Further, the detailed design of the MOS/Deep Space Network (DSN) software is directly derived from the Lunar Orbiter mission-dependent software, which is, in turn, based on the Mariner software.

Based on JPL Mariner experience, a preliminary definition of the software has been developed for use in analyzing trends manifested in the test data to identify incipient failures or performance degradations in Voyager spacecraft hardware, before they occur. The team approach that has been employed in developing these programs is summarized in Figure 3.6-1. The basic elements making up the computer software system are shown on Figure 3.6-2.

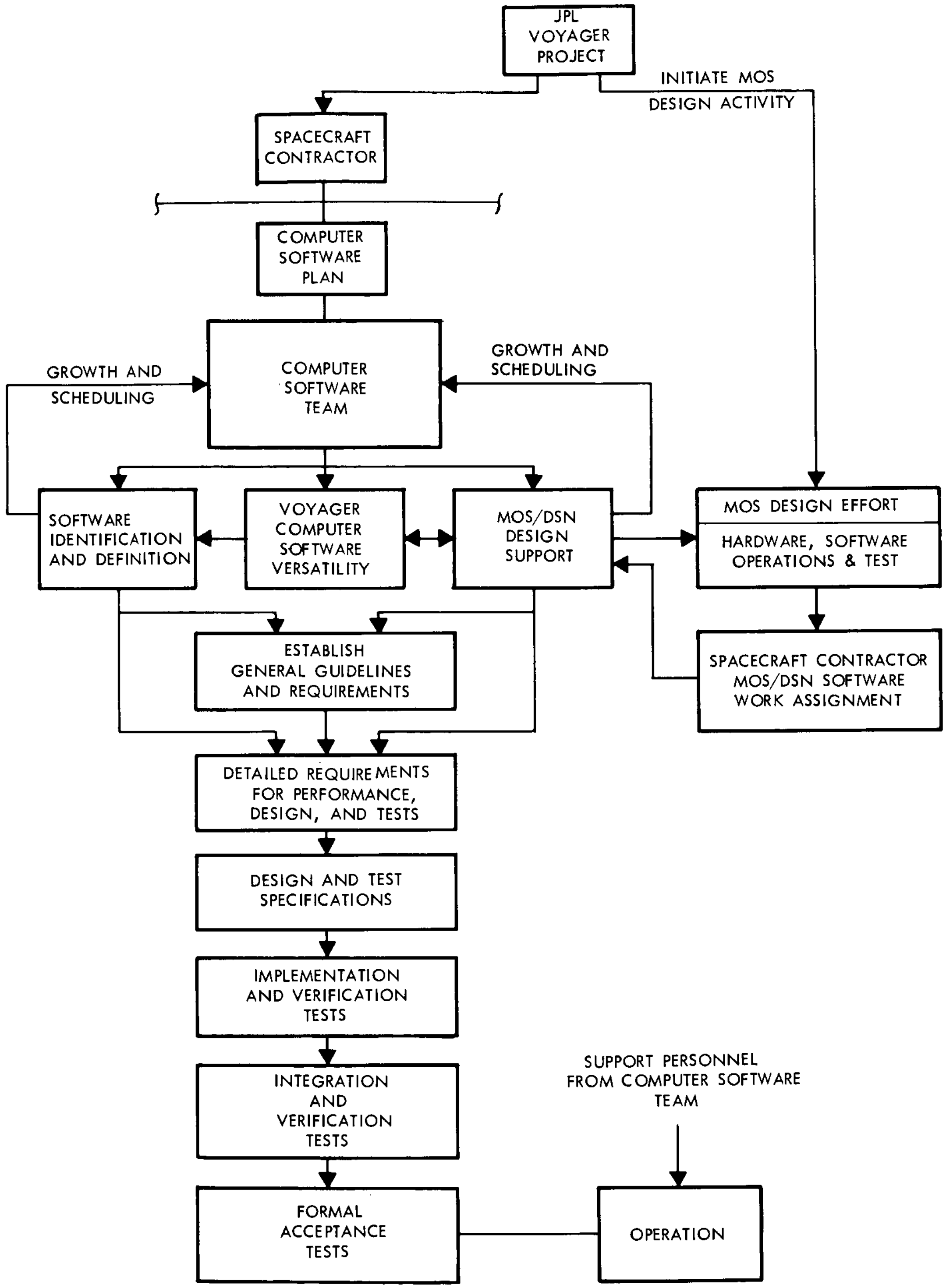
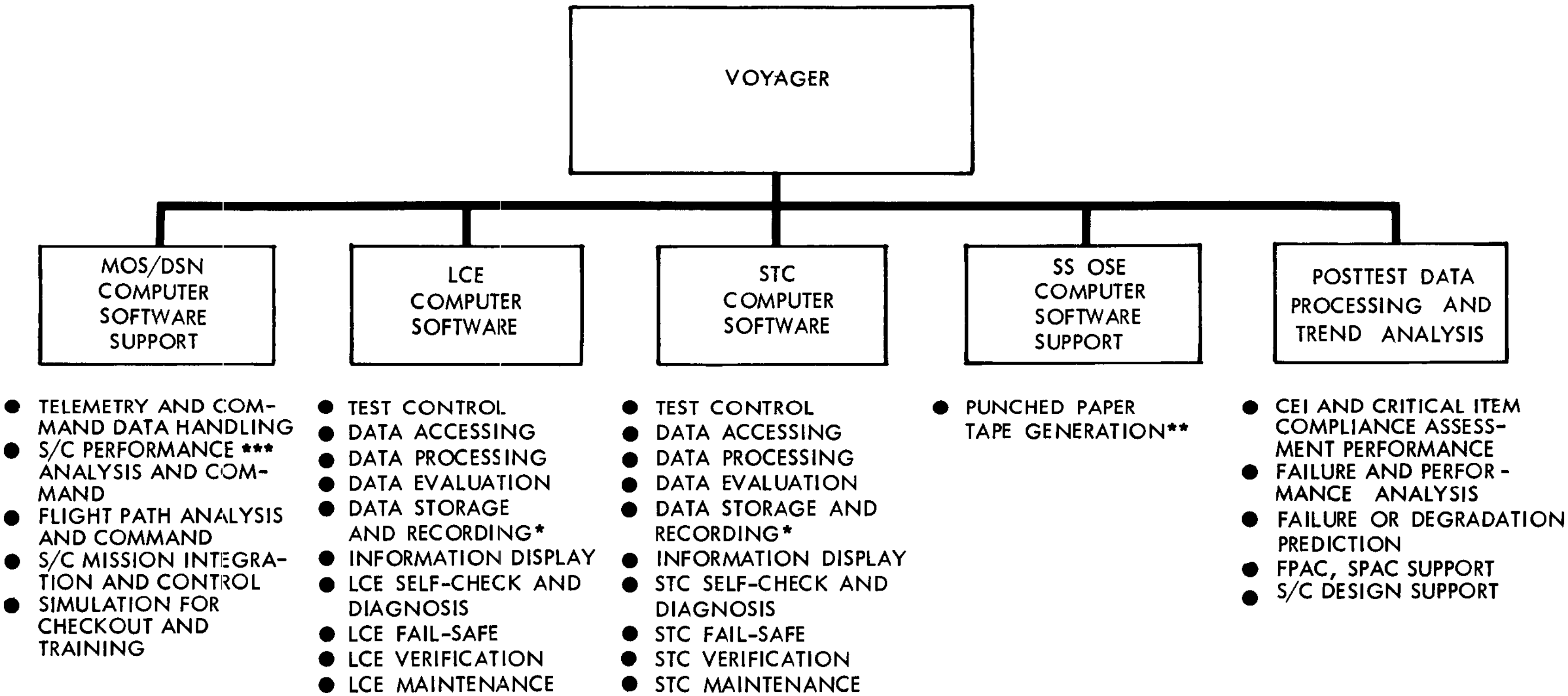


Figure 3.6-1: Computer Software Development Plan



* RECORDED TREND DATA ACCUMULATED AND SUPPLIED FOR TREND ANALYSIS
** RECORDED TREND DATA ACCUMULATED AND SUPPLIED FROM SS OSE
*** RECORDED DATA WILL BE USED IN UPGRADING TREND ANALYSIS FOR SUBSEQUENT FLIGHTS

Figure 3.6-2: Areas Requiring Computer Software Design Support And Development

3.6.2 Applicable Documents

1) EPD 125, "Programming Standards for SFOF User Programs," July 15, 1965.
2) EPD 23, "SFOF Data Processing System," July 20, 1964.
3) NASA Document CM001BB001-1B, "Preliminary Voyager Configuration Management Manual," November 12, 1965.
4) TR-32-223, "Space Trajectories Program for the IBM 7090 Computer, Revision 1."
5) ED 171, "SFOF User Area Capability," July 1, 1963.
6) ED 199, "Users Guide for JPTRAJ," January 4, 1964.
7) EPD 317, "DSN Software Integration Plan for Lunar Orbiter," September 20, 1965.

3.6.3 Design Constraints and Requirements

The design constraints and requirements of the Voyager spacecraft OSE software are presented below. These design constraints and requirements are derived from the applicable documents above and from the Voyager Spacecraft System functional requirements represented in the system engineering functional flow plans.

3.6.3.1 Constraints

The constraints that govern these software items are those required to ensure maximum compatibility with operating environments and compliance with JPL direction. The specific constraints of the major subsets of OSE software are discussed below.

1) Mission-dependent software:

   a) Format compatibility with MOS/DSN (applicable to both data and commands);

   b) Modular structure and linkage compatibility with mission-independent programs to SFOF (applicable to 7094-44 programs at SFOF).

2) LCE computer software:

   a) The limited number of measurements available through telemetry and umbilicals;

   b) Data format compatibility with DSIF 71.

3) STC computer software--STC operational usage constrains the program structure to accommodate computer usage to support STC operation in manual and semiautomatic modes.

4) Post test data-processing software--The post test data-processing software used in the SFOF must conform to the operating procedures and computing system structure of SFOF.

3.6.3.2 Requirements

General Requirements--General requirements that apply to all software in this general report govern reliability and safety, ease of checkout, modular design, self-checking and error-diagnostic abilities, program languages, and documentation.

Reliability and Safety--No operation of the computer programs can be permitted to cause a situation to exist that is hazardous to personnel or potentially damaging to the spacecraft. Fail-safe devices and protective circuits in the OSE and spacecraft do not relieve the software of this requirement.

Ease of Checkout--

1) Independent checkout of modules with simulated interfaces;

2) Independent checkout of subprograms;

3) Breakpoints in subprograms to permit checkout as part of a program;

4) Capability to check out programs with simulated interfaces;

5) Capability to check out programs in the actual operating environments with both actual and simulated test articles.

Modular Design--The software will be designed in modules that can be checked out individually at the subroutine level before being integrated into higher level programs and into processing systems.

Self-Checking and Error Diagnosis--Capability for every program system and program to run a self-test routine in which a test case with pre-established results confirms that the program and the computer are operating correctly.

Programming Language--Only proven compilers and assembly programs that are in current general use will be used in writing these programs.

Documentation--Compliance with the requirements set forth in Exhibit XIX of the "Voyager Configuration Management Manual." Additional documentation standards require that detailed flow charts of the program must be produced. These logical flow charts are to contain sufficient detail to indicate every logical branch in the program modules. Source-program listings are to be annotated in such a way that there is an easy correlation between the compiler listings and the corresponding logical diagram.

MOS/DSN Computer Software Requirements--These requirements are listed below:

1) Support two spacecraft simultaneously;
2) Perform all mission-dependent telemetry and command data handling functions;
3) Perform the Voyager spacecraft-unique flight-path analysis and command functions (particularly those related to Voyager spacecraft weight, center of gravity, and engine parameters).
4) Perform the computations required to support analysis of spacecraft performance and spacecraft subsystem performance and characteristics.
5) Perform the computations required to maintain and update weight, center of gravity, fuel remaining, and engine performance parameters.
6) Provide a capability to receive off-line or manually prepared performance and flight-path data.
7) Provide computational support to mission integration and control.

LCE/STC Computer Software Requirements--The design of the LCE and STC software takes maximum advantage of LCE and STC design commonality by using a common program control structure and common program elements. Differences in functional operation are satisfied by interchanging or adding program modules rather than supplying two different programs. The following is a list of functional requirements for the LCE and STC.

1) A common executive control routine is required which provides:
    a) Control input from test director's display and control console to exercise various program options;

b) Alarm detection and subsequent spacecraft conditioning to prevent equipment damage;

c) Test selection and sequence scheduling;

d) Test data processing for on-line evaluation, display, and recording;

e) Initiation of self-check routines for verification of test equipment without interruption of spacecraft testing;

f) Sufficient programmed safeguards to prevent the occurrence of damage due to improper test sequences;

g) A test log of pertinent test information.

2) Prevention of inadvertent alteration of the program structure by any data entry routine.

3) In the LCE, to route capsule system or Science Subsystem data to capsule LCE or Science Subsystem OSE.

4) In the LCE, compatible interface with T-countdown timing and shutdown signals provided by the launch vehicle systems; also to provide "no-go" signals to Launch Vehicle ESE when spacecraft "no-go" conditions are recognized.

5) In the LCE, to receive and evaluate telemetry data from DSIF 71 to verify compatibility of spacecraft with the MOS during launch complex tests.

Subsystem OSE Computer Software Requirements--

1) Generate punched paper tape for use on the SS OSE.

2) The requirement for data processing and analytic computations, including trend, of SS OSE generated data is satisfied by the post test data processing and analysis software (see below).

Post test data processing and analysis software requirements--Functional requirements are data processing and analytic computational support to:

1) Trend analysis;
2) Contract End Item (CEI) and critical-item specification compliance assessment;
3) Failure analysis;
4) Performance analysis.

Performance Requirements--

1) Accept the data in the various recording modes of SS OSE, STC, and LCE.
2) Time correlate and perform combinational calculations of data from the OSE and major test facilities (e.g., thermal-vacuum test chamber).
3) Store data in and retrieve data from the voluminous files required for trend historical data, instrument calibration data, and specification requirements and tolerance data.
4) Control and operate variable sequences of modules and programs.

Trend Analysis Software Requirements--The basic requirements for the trend analysis system are presented in Volume C of the Voyager Task A Report, "Design for Operational Support Equipment."

Requirements for the computational software to support trend analysis are amplified as follows.

1) Computational subroutines and programs to support trend analysis are required to provide rapid access to correlated listing of any subset of Voyager test measurements.
2) Computer programs must not require usage of the STC or LCE computers.

3) Filekeeping system must provide automatic access to all trend-significant data taken through the telemetry system.

4) Filekeeping system must provide automatic access to the index of all historical test data.

3.6.4 Functional Description

The following functional descriptions present the salient features of the computer software programs.

3.6.4.1 MOS/DSN Software

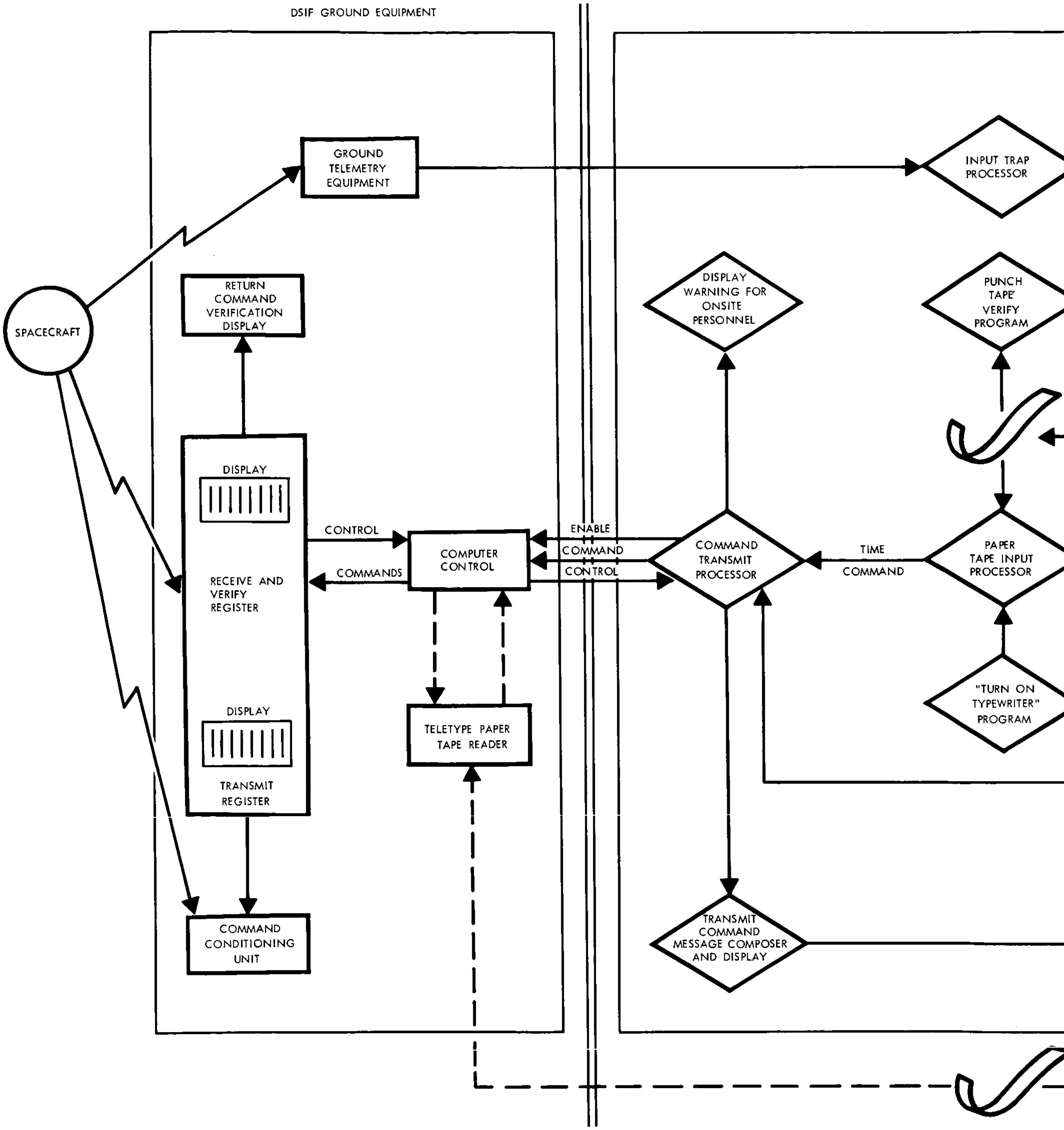
The spacecraft mission-dependent computer software for the MOS/DSN is divided into the following categories:

1) Telemetry and command data handling (TCD);
2) Flight-path analysis and command (FPAC);
3) Spacecraft performance analysis and command (SPAC);
4) Mission integration and control (MIC).

Telemetry and Command Data Handling (TCD)--The TCD computer software consists of an integrated set of computer programs written for the DSIF SDS 920 (or 930) computer. The design of the TCD computer software is patterned closely after Lunar Orbiter TCD programs, but will be updated to reflect the planned capabilities of the DSN.

Figure 3.6-3 indicates the telemetry and command data flow within the DSIF.

The major software elements are indicated and are described as follows:
Command Processor--The command processor provides the capability for receiving, verifying, displaying, storing, and controlling transmission


DSIF GROUND EQUIPMENT
GROUND TELEMETRY EQUIPMENT
INPUT TRAP PROCESSOR
SPACECRAFT
RETURN COMMAND VERIFICATION DISPLAY
DISPLAY WARNING FOR ONSITE PERSONNEL
PUNCH TAPE VERIFY PROGRAM
DISPLAY
RECEIVE AND VERIFY REGISTER
CONTROL
COMMANDS
COMPUTER CONTROL
ENABLE
COMMAND
CONTROL
COMMAND TRANSMIT PROCESSOR
TIME
COMMAND
PAPER TAPE INPUT PROCESSOR
"TURN ON TYPEWRITER" PROGRAM
DISPLAY
TRANSMIT REGISTER
TELETYPE PAPER TAPE READER
COMMAND CONDITIONING UNIT
TRANSMIT COMMAND MESSAGE COMPOSER AND DISPLAY

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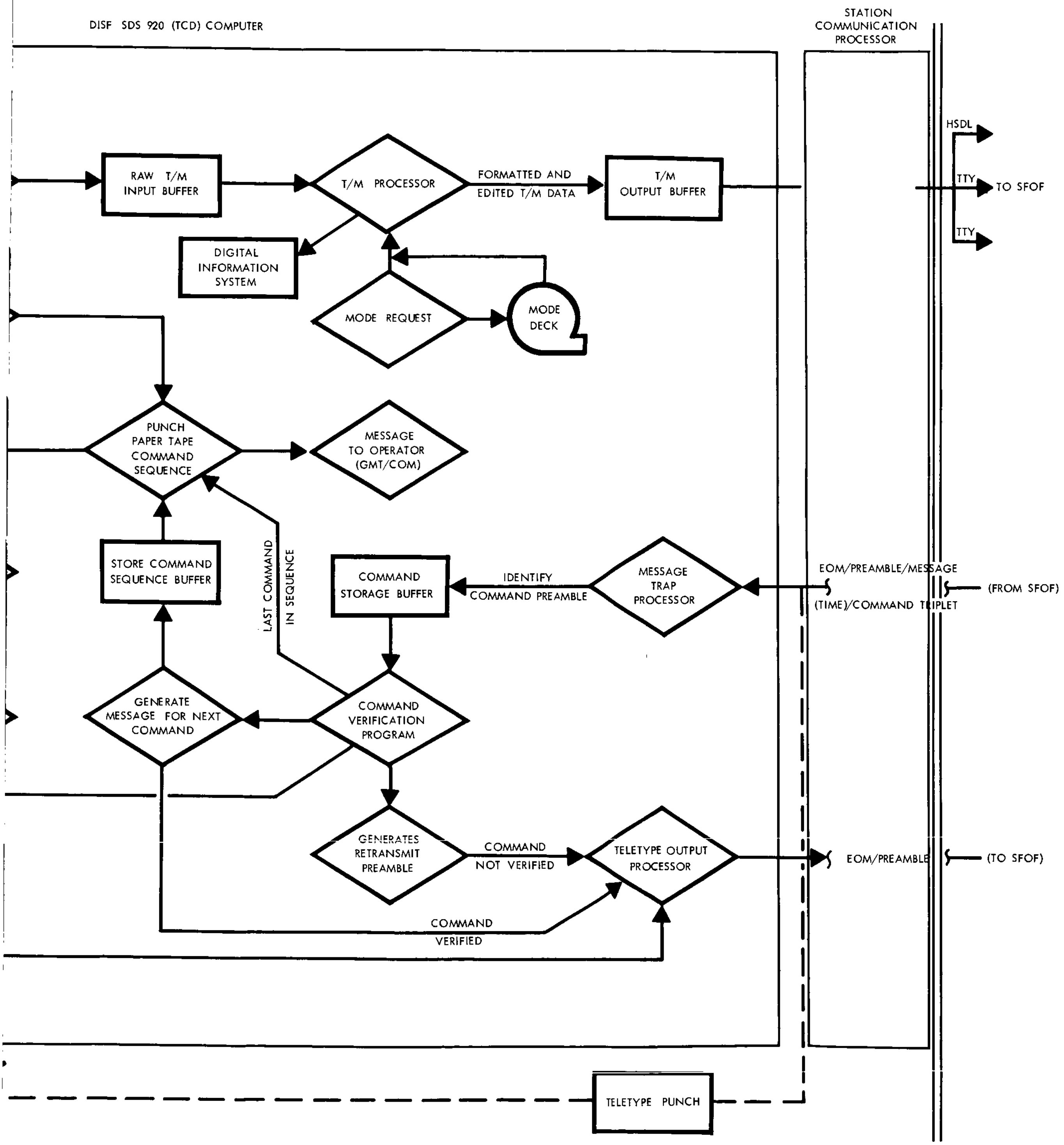


Figure 3.6-3: DSIF TELEMETRY AND COMMAND DATA FLOW

2

of commands from the SFOF to the spacecraft via Voyager ground equipment at the DSIF. The command processor consists of two major routines.

- Command Verification Processor--The command verification processor has the capability to accept a string of command triplets transmitted from the SFOF. This processor constantly looks at incoming teletype messages coming over the teletype lines. If the command message is verified, the on-site computer looks for the next command triplet from the SFOF. If the command was not verified, the same command triplet is retransmitted from the SFOF. This sequence of events continues until the last command triplet has been verified.
- Command Transmission Processor--At a preset time, the command transmission processor alerts the DSIF operator to indicate the time in minutes which will elapse before a command string will be sent to the Voyager spacecraft. After the command has been transmitted, verified, and enabled by the ground equipment, a signal requesting the next command is sent back to the SDS 920. At this time the SDS 920 issues an administrative message, which is sent to the SFOF reporting that the commands have been transferred from the SDS to the command transmission equipment. The above command transfer control, printout, and SFOF notification is repeated until the complete command sequence is transmitted, verified, and stored in the spacecraft programmer.

Telemetry Data Processor--The telemetry data processor buffers, formats, and edits the telemetry data stream as it is received from the on-site ground equipment and controls the routing of the telemetry data to the station communication processor (SCP). To accomplish the above, the telemetry data processor includes two major routines.

- Input Trap Processor--The input trap processor accepts data from the telemetry decommutator and transfers it into a memory buffer, controlled by the external interrupts, frame sync, and word sync generated by the ground equipment. This routine also identifies each of the frames as they are stored in an input buffer.
- Telemetry Processor--The telemetry processor formats and edits, prior to retransmission, raw telemetry data from the input buffer. In some modes, the teletype lines, at a 30 bps rate, are not capable of transmitting all the raw telemetry data in real time. Therefore, several edit modes accommodate the transmission rates. The telemetry edit modes also select various priority measurements and combinations of measurements for priority transmission to the SFOF or to the digital information system (DIS), at the DSIF during the various mission phases.

Flight Path Analysis and Command (FPAC)--The FPAC computer software provides prediction positions and velocities and maneuver sequences for the spacecraft trajectories. The software will be programmed for the IBM 7094 and associated peripheral equipment. Following is a brief description of the major functions of the FPAC computer software:

1) Computes the spacecraft trans-Mars trajectory orbital elements and associated errors based on nominal trajectory information and DSIF tracking data;
2) Calculates predicted spacecraft positions and velocity;

3) Compute optimum Mars orbit injection conditions to minimize injection velocity requirements for given conditions;
4) Perform a preliminary trajectory search using fast and approximate conic computation;
5) Perform a precise midcourse trajectory search to find a trajectory that satisfies the optimum Mars-orbit-injection conditions;
6) Predict the spacecraft maneuver errors, combining these errors with the tracking errors, and mapping the total error volume to Mars orbit injection;
7) Compute the pitch, yaw, and roll commands to effect the required midcourse velocity changes, taking into account the attitude constraints of the vehicle;
8) Predict the doppler change that should occur for a successful midcourse maneuver;
9) Reestablish the spacecraft trajectory, after midcourse correction, using the old spacecraft position and new DSIF tracking data;
10) Compute the Mars-orbit-injection maneuvers that ensure the minimum-injection-velocity maneuver applicable to scientific objectives;
11) Compute the thrust orientation and duration required to achieve the selected injection maneuver, taking into account the effects of finite thrust time;
12) Compute the pitch, yaw, and roll maneuver required to effect the computed thrust orientation, taking into account the spacecraft maneuver constraints;
13) Predict the spacecraft maneuver errors, combining these with doppler prediction errors, and mapping the errors through nominal transfer to science experiment time;

14) Reestablish the vehicle trajectory after orbit injection;

15) Determine the significant Mars harmonic coefficients;

16) Perform a transfer orbit search, using approximate but fast orbit computation procedures, to find an orbit from which specified scientific objectives can be achieved.

Spacecraft Performance Analysis and Command (SPAC)--The SPAC computer software provides monitoring, controlling, and trend analysis for each spacecraft subsystem. Included in the SPAC computer software are the following computing elements:

1) Electrical power and energy management;
2) Thermal management;
3) Gas budget and vehicle dynamics;
4) Signal margin prediction;
5) Star identification;
6) Velocity control.

These elements perform the following functions:

1) Perform the analyses required to assess subsystem performance, and provide status information to the user area;
2) Predict future subsystem capabilities and status as a function of mission time and event sequence (trend analysis);
3) Format output data in the user area for specified display devices and in specified formats;
4) Accept in accordance with preestablished priority call-up from the user area input-output consoles;
5) Store on disk or magnetic tape specified program outputs for call-up by other programs;

6) Accept manually input data from punched cards or message composers in lieu of, or in addition to, data from the master data files or user program files.

The MIC software has the capability to:

1) Correlate the spacecraft clock time with Greenwich Mean Time and detect errors in timing that may have occurred in the spacecraft as required;

2) Transmit commands to the spacecraft in an exact format and in a planned sequence. (Inputs for commands will be received from both FPAC and SPAC and will be transmitted by the command coordinator.);

3) Convert the command parameters received in engineering units to command word format and provide the command sequencing requested by the command coordinator;

4) Update the mission countdown to account for variation in trajectory parameters by providing the SFOF with software that will list both past and future events within a 24-hour period;

5) Provide the user programs with all items of parametric data for their operation through common environment programs consisting of three routines:

   a) A utility program, operating independently of other user programs, to perform initial loading and housekeeping functions,

   b) An input program to be employed by a user program in reading data from the file,

   c) A routine for taking parametric results generated by a user program and storing them for entry in the common environment.

3.6.4.2 LCE/STC Computer Software Functional Description

The computer software functional requirements for the LCE and STC are basically the same. Because of this, a common organizational structure is used. Specific functions are accomplished by supplying the appropriate modules.

The functional description of the STC computer software is given below. This is followed by a description of the function changes necessary for LCE operations.

Figure 3.6-4 illustrates the major functional elements of the STC computer program and is described, by elements, as follows.

STC Executive Control Routine (ECR)--This routine provides overall program integration and control. Initialization of the ECR is accomplished from the test director display and control console (TDCC). Upon initialization, the TDCC function switches are scanned. At this time the TDCC operator can set up or execute the following program options:

1) Select one of the console input devices;
2) Select specific tests or test sequences;
3) Select and request data for display;
4) Select test mode;
5) Initiate tests.

The ECR also provides for alarm sensing via the priority-interrupt channels. If an alarm is sensed, the ECR branches to a fail-safe routine for further action.

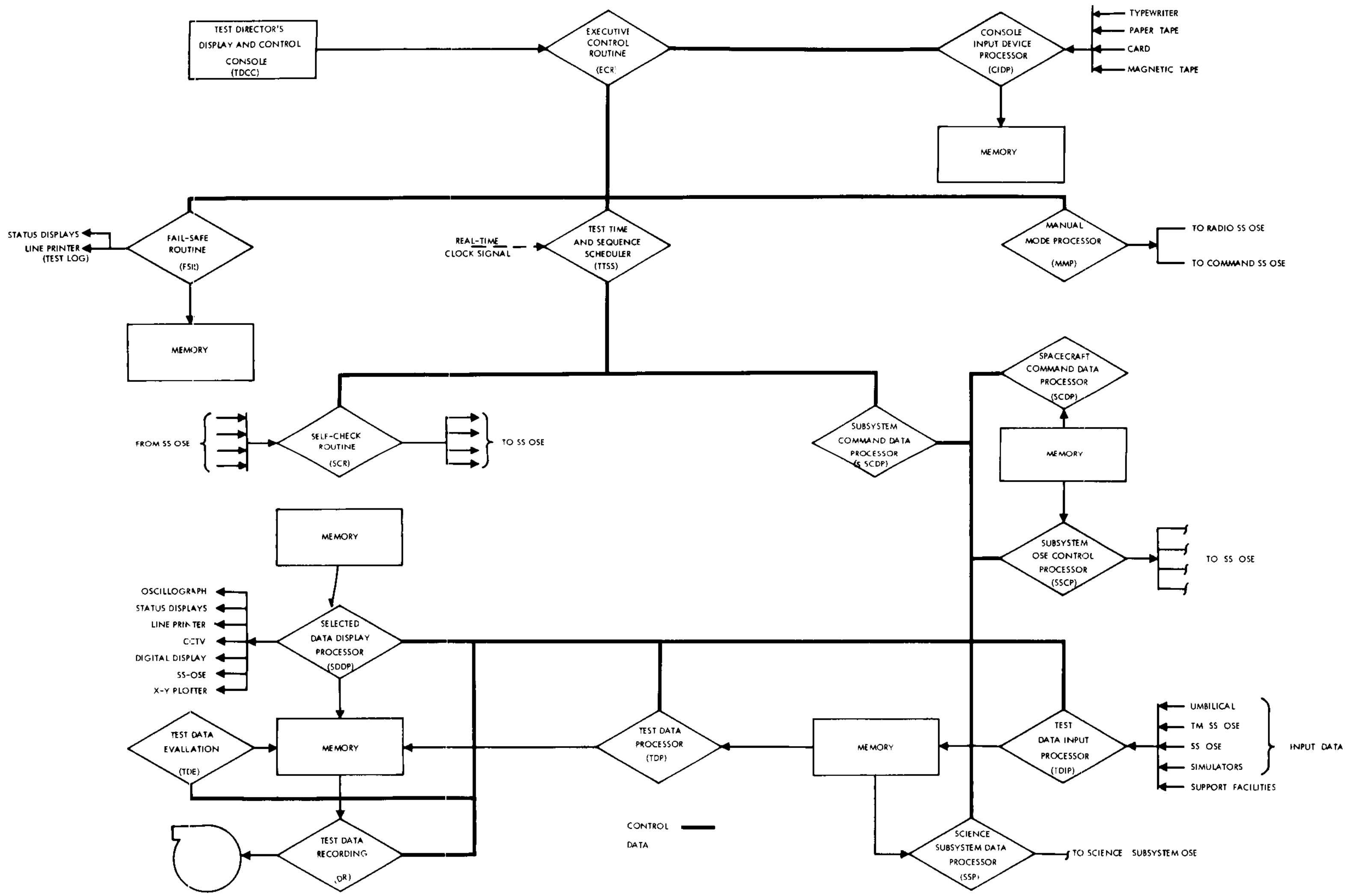


Figure 3.6-4: STC Computer Software Control And Data Flow Diagram

Specific tests and test sequences are selected by data entry into the computer using typewriter, cards, or tape. The selected tests and test sequences are noted on the test log.

Selection of the test mode determines whether the program will run automatically, semiautomatically, or manually. Initiation of test begins actual test operation by entry into the test time and sequence schedule routine.

Fail-Safe Routine (FSR)--The fail-safe routine provides controlled "shutdown" of the test operation in the event of a sensed alarm. FSR immediately scans all critical information input lines and stores the data in memory. It then initiates commands to the various subsystem test sets to condition these units to a fail-safe test status. FSR then outputs to the status displays and prints out on the test log all critical information. Upon completion of printout, the FSR returns to the ECR for further instructions.

Console Input Device Processor (CIDP)--The test director display and control console operator, by way of function switches and via the ECR, can select one of four devices to enter information into the computer. These devices are the typewriter, card reader, paper-tape reader, and the magnetic-tape unit.

The CIDP is designed to prevent accidental alteration of the program structure by erroneous input information. All data to be entered is preceded by a code word and a memory location. The code word corresponds to assigned memory blocks. If the specified memory location is not within this block, the input process is halted and an error is flagged on the typewriter and test log. The typewriter can also be used to request or provide information display.

Test Time and Sequence Scheduler (TTSS)--The TTSS is entered via the ECR upon selection of the "Initiate Test" switch. The TTSS operates in two modes, automatic and semiautomatic, depending on the mode selection switch. In the automatic mode, all tests and test sequences are automatically executed. In the semiautomatic mode the test director display and control console operator (TDCC) can single-step a test when a single-test step is meaningful. In both modes, the TDCC operator can "hold" the program, recycle tests, or initialize the program to return to the ECR for further instruction. In "hold," the TDCC typewriter can be used for data display requests.

A real-time clock provides the time base for scheduling test events. A basic time interval is established and an interrupt occurs, signaling the TTSS that this interval has elapsed. Items scheduled on this time-base, according to a stored test, time, and sequence function table, are as follows: 1) Selected spacecraft command data processed for output to the radio or command SSTS; 2) SSTS test and sequencing commands processed for output; 3) Test data input processing; 4) Test data recording; 5) Test data evaluation; 6) Selected data processing for display; 7) Self-check.

Upon completion of test scheduling, the TTSS returns to the ECR for further instruction.

STC Test Data Processor (STCDP)--The test data processor consists of the following.

Spacecraft Command Data Processor (SCDP)--Under TTSS control, the SCDP selects, formats, and processes spacecraft command data for output to the radio SSTS or the command SSTS.

SSTS Control Processor (SSCP)--Under the TTSS control, the SSCP issues test identification and test sequence control commands to the SSTS.

Test Data Input Processor (TDIP)--Under control of the TTSS, the TDIP inputs, to assigned memory locations, data from the spacecraft via the SSTS and spacecraft umbilical.

Space Science OSE Processor (SSP)--Under control of the TTSS, the SSP processes Science Subsystem input data for transmittal to the Space Science OSE.

Test Data Processor (TDP)--Under control of the TTSS, the TDP processes test data according to type by identifying, time tagging, and converting to engineering units. Processed data are stored for future use in assigned memory locations.

Test Data Recorder (TDR)--Under control of the TTSS, the TDR selects and formats test data for recording on magnetic tape for post-test analysis.

Test Data Evaluater (TDE)--Under control of the TTSS, the TDE evaluates selected test data as follows:

1) Limit checking;
2) Data change detection;
3) Event monitoring and time correlation.

Out-of-tolerance data are flagged for display at the appropriate SSTS and at the test director's display and control console (TDCC).

Selected Data Display Processor (SDDP)--Under control of the TTSS and the console switches, the SDDP processes selected data for printing, plotting, digital display, and SSTS adapter display. The SDDP is responsible for keeping the test log (see Section 3.1.5.2).

Self-Check Routine (SCR)--The SCR operates under control of the TTSS or may be called by the TDCC function switches via the ECR. The SCR initiates self-check commands to selected test sets and monitors results. It also performs an internal self-check. If a fault is detected, it is displayed on the TDCC and the appropriate SSTS.

Manual Mode Processor (MMP)--Manual mode operation is essentially a monitoring mode. The SSTS are autonomous in this mode, but are monitored for test sequencing; an inhibit command is generated if a damaging sequence is sensed. An indicator at the TDCC and the appropriate SSTS displays inhibit status.

LCE Computer Software Functional Description--To adapt the STC to LCE operation requires program module changes and additions as follows.

1) The manual mode operation is deleted.
2) Semiautomatic mode is inhibited for countdown operations.
3) The test, time, and sequence schedule table is changed to reflect the LCE test schedule.
4) DSIF 71 data processor and evaluator (DSNE) is incorporated into the software. This is used to compare DSIF 71 data with umbilical and SSTS telemetry data to verify spacecraft-DSIF compatibility and operational readiness of the DSN.
5) The real-time clock is slaved to launch vehicle countdown time.

6) Data processing modules are incorporated for relaying data to the capsule SS OSE.

7) No-go and shutdown processor is incorporated to generate go/no-go signals for the launch vehicle ESE and accept shutdown signals from launch vehicle ESE and then condition spacecraft to a safe status.

3.6.4.3 Post Test Data Processing Functional Description

Figure 3.6-5 depicts a design concept for a test-data-reduction computing system. The following paragraphs summarize the functional relationships of the various elements shown in the figure.

Overall control of the computing system is maintained by the system executive. The data-reduction monitor provides an interface with the system executive making a data-reduction job appear to the system executive like all other jobs, and in addition maintaining control through the system executive of the data-reduction sequences.

Two levels of priorities may be provided: system level, the relative priority of test data; and reduction priority, the relative priority between data processing requests. The former priority is determined by the system executive and the latter by the data processing monitor.

Data-reduction support files are maintained automatically, and contain controls for the data reduction monitor; processed data in display format, for almost immediate output to display devices; processing control data such as calibrations, constants, and tables; checked out data-reduction programs; and indexes to the location of all active data that has been entered to the system for processing, storage, or display.

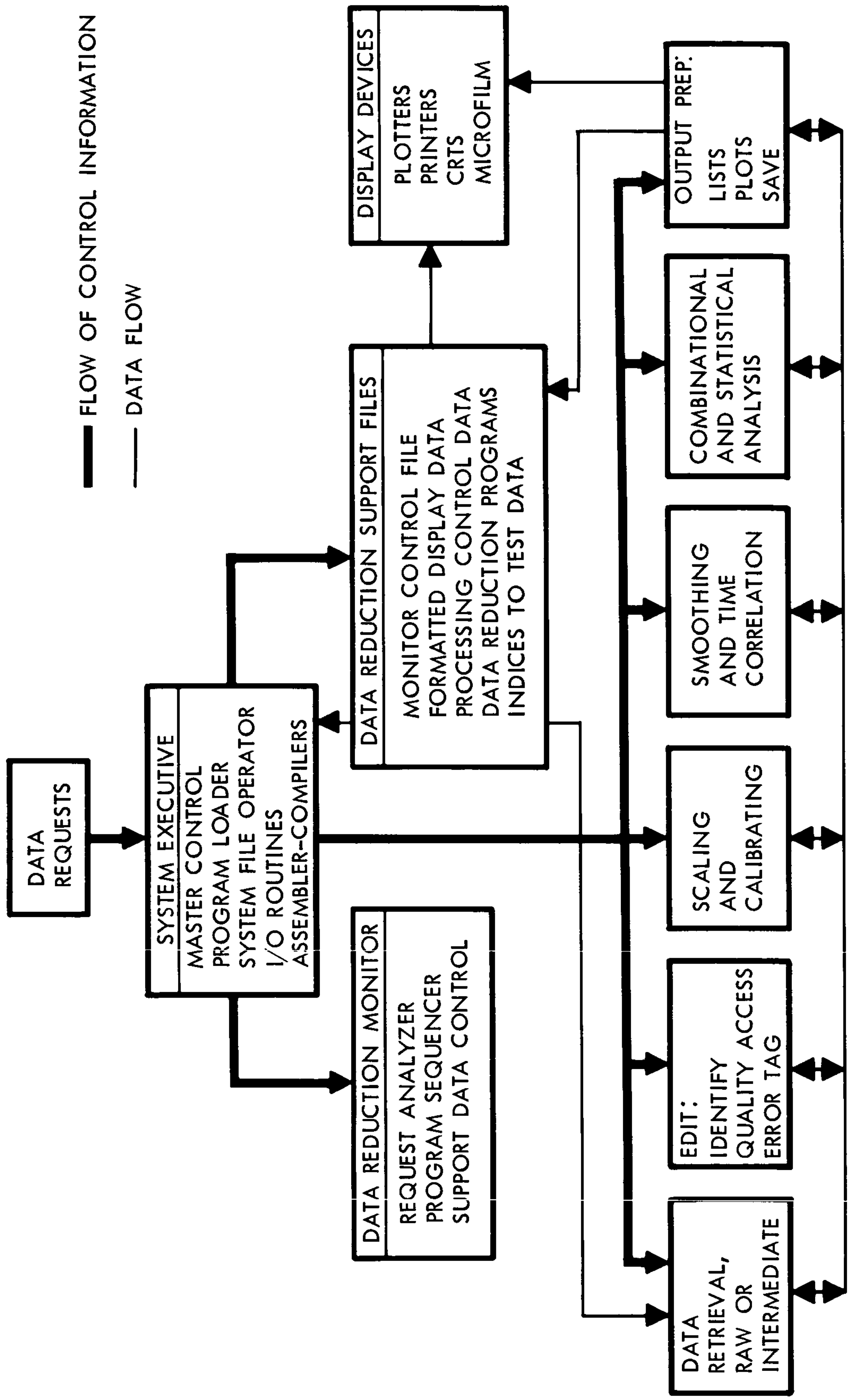


Figure 3.6-5: Posttest Data Processing Design Concept

The system provides for all of the normal processes required in the computer reduction of test data, from raw data retrieval through final display; for saving the results of any process for further processing or display at a later time; and for updating support files.

On recognition of a data-reduction request, the system executive passes control information to the data-reduction monitor and allows it to operate. The data-reduction-monitor elements shown provide the system executive with controls needed for loading the programs and support data needed to process the data requests.

A typical processing sequence might start with retrieval of data in raw form or such intermediate formats as edited, scaled and calibrated, or smoothed. Many culminate in final or intermediate results for either immediate display or file updating. The structure of a data-reduction program is described below.

<u>Data-Reduction Program Structure</u>--Data-reduction programs for operation within the framework described above are built up from generalized computation routines, each of which has been checked out individually, prior to integration, in an environment that simulates the data-reduction system. Generalized input and output (retrieval and filing) and display (list and plot) programs provide an interface between computation routines and input and output and display equipment. Computation routines vary in complexity from those required for simple linear calibration and scaling to those required for digital filtering and integration.

Figure 3.6-6 represents a typical data-reduction program. It is composed of various control blocks, and calls to input, calculation, and output programs. Communication between input, computation, and output is maintained through the following control blocks

Variable Control Blocks (VCB)--A VCB is a setup for each input, computed, or output variable and contains these variables: value; quality code; type, integer or floating point; and word size. Input and output variables also contain a pointer to a sampling sequence table.

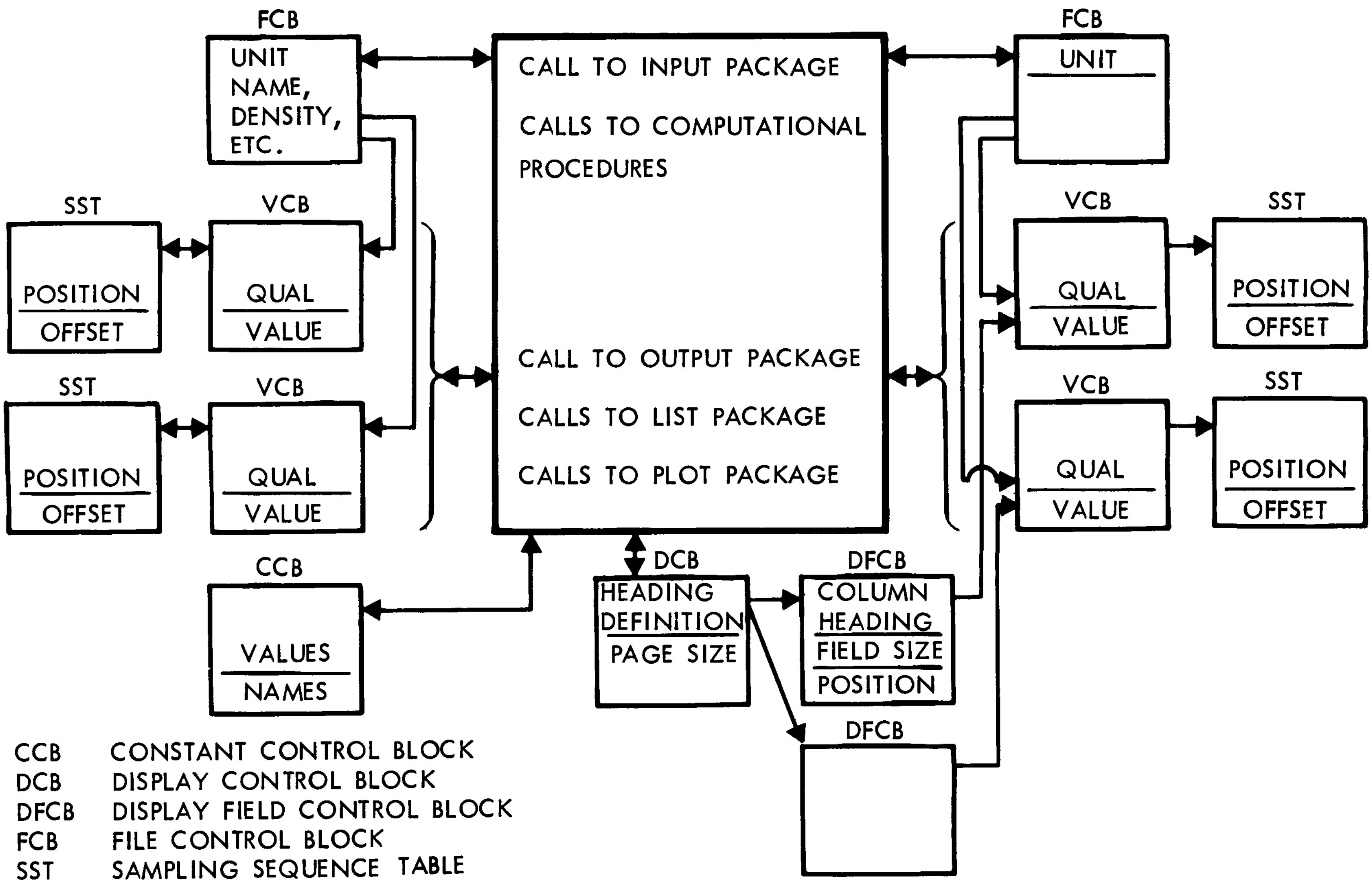
Sampling Sequence Tables (SST)--An SST contains a variable identification code and an offset value, relative to frame time, for each position of a variable within a data frame.

File Control Blocks (FCB)--An FCB is associated with each data source or destination file and contains such file identification as reel numbers and recording density. An FCB also contains pointers to the VCB's of all variables to be retrieved or stored on file.

Display Control Blocks (DCB)--DCB's contain the information required to define each display. A list control block, for example, will contain page width and height and display heading information. DCB's point to a display field-control block and to the VCB for each variable to be displayed.

Display Field Control Block (DFCB)--DFCB's define field-width and column-headings information for each displayed variable.

Constant Control Block (CCB)--A CCB is maintained for each constant or table required for computation. CCB's contain the names and values of each constant or table.



CCB	CONSTANT CONTROL BLOCK
DCB	DISPLAY CONTROL BLOCK
DFCB	DISPLAY FIELD CONTROL BLOCK
FCB	FILE CONTROL BLOCK
SST	SAMPLING SEQUENCE TABLE
VCB	VARIABLE CONTROL BLOCK

Figure 3.6-6: Data Reduction Program Structure

The input package retrieves data from the input file designated by the file-control block, and fills the VCB's and SST's. Computation programs retrieve required data from the VCB's, using standard subroutines for manipulating quality codes and performing similar functions. Computation results are stored in output VCB's. The output programs retrieve output data from the output VCB's. Display definition is supplied by the DCB's and DFCB's.

3.6.4.4 Trend Analysis Software

The computer applications to support trend analysis are designed as a subset of the post-test data processing software for the Voyager Spacecraft Project. The spacecraft performance analysis programs are augmented by special trend-analysis functions that analysis personnel can apply selectively to the outputs of the performance analysis routines. The major categories of special trend-analysis functions are as follows:

1) Noise removal routines--digital filter and curve-fitting processes for extracting or removing higher frequency, random components from a data train;
2) Very-low-frequency curve-fitting techniques for extracting the long-term trends from data records;
3) Routines, such as Fourier and power-spectral-density analysis, for determining the cyclic component of a data record;
4) Statistical analysis computations consisting of parametric and nonparametric statistical processes for determining the mean, median, range, and standard deviation of a data record and consisting of variance and covariance techniques for evaluating the interaction among measured phenomena;

5) The executive (or master control program) that manages the linkage and input-output functions of the computational routines in such a way that an analyst can select virtually any sensible sequence of routines.

The index for trend data contains five major categories under which data are stored:

1) Measurement and part number identification;
2) Test identification;
3) Applied environmental stress nomenclature;
4) Time (day, hour, minute, etc.);
5) Trend parameter identification.

The first four use the standard nomenclature and numerical identification used throughout the Voyager program for those categories of data. The last (trend parameter identification) uses nomenclature and numerics specially developed for the trend-analysis system. Major subcategories of trend-parameter identification are accumulated running time, accumulated stress cycles, failure mode identification, and the nomenclature of the analysis techniques (i.e., the statistical or mathematical process used on the data). Access to a specific data item is accomplished by querying the system with two or more identification items in at least two index categories.

The above functional elements of the trend-analysis system are defined to have modular construction that conforms to the requirements of the posttest data processing system and, when operating at SFOF, with SPAC.

3.6.5 Interfaces

3.6.5.1 MOS/DSN

Interfaces between the mission-dependent computer programs in the SFOF are with those JPL programs that are basically mission independent. These are specified in JPL Document EPD-125, "Programming Standards for SFOF User Programs," July 15, 1965. Functional interfaces at the DSIF are between the TCD computer and the spacecraft MDE, the site communications processor (SCP), and the digital instrumentation system (DIS). These interfaces are described in EPD-283, "Planned Capabilities of the DSN for Voyager," July 15, 1965.

3.6.5.2 LCE/STC

Functional interfaces with the LCE and STC computer programs are between the LCE/STC computer and associated equipment. These interfaces are discussed in Sections 3.1 and 3.2.

3.6.6 Reliability and Safety

Section 3.6.3.2 establishes software reliability and safety requirements. Major emphasis is given to STC/LCE and MOS/DSN software. Design and implementation approaches are such as to maximize the reliability and safety. The modularity principles that have been selected as requirements are intended to facilitate checkout and yield improved confidence in the software.

3.6.7 Test

All computer software will be tested to performance specifications at the subroutine levels and all levels thereafter, before integration of the program into its operating environment. Integration tests to establish compatibility and overall functional design performance will be performed after lower levels of checkout.

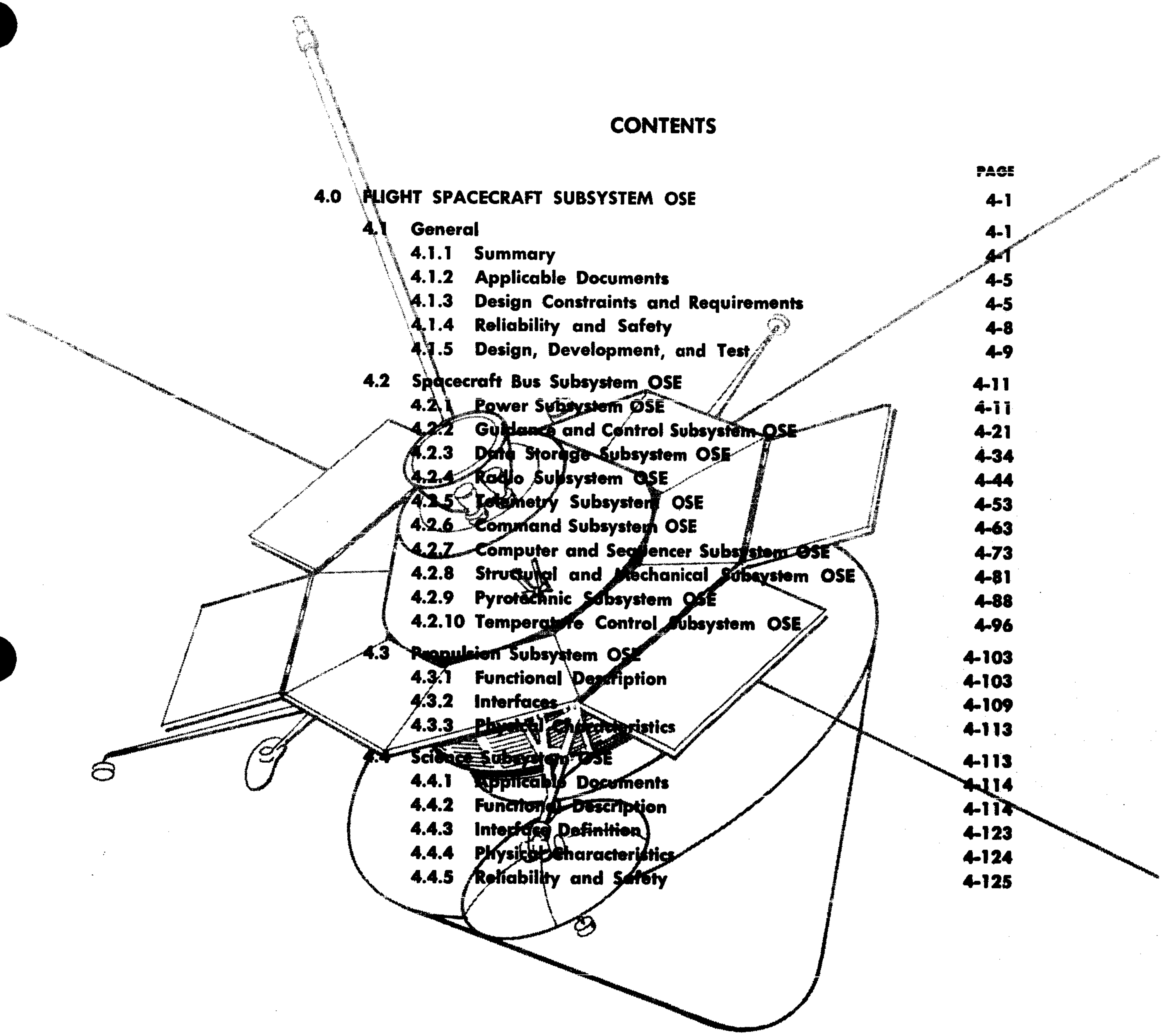
3.6.7.1 MOS/DSN

The above-outlined checkout plan applies to all programs. Programs designed for use on the DSN computer systems will be verified at the spacecraft contractor's facility, configured to simulate the SFOF interfaces prior to integration into the DSN system. All TCD (DSIF) computer programs will be similarly checked out on an SDS 920 (930) computer with DSIF interfaces simulated. Integration of the computer programs into the DSN will be subjected to JPL integration tests (example reference, EDP-317).

3.6.7.2 LCE/STC

The LCE/STC computer programs will be checked out on an SDS 920 (or 930) computer prior to integration into the LCE/STC operating environment. The computer interfaces will be simulated to verify, independently, the functional performance of the computer software. Similarly, computer programs will be written to aid LCE/STC hardware checkout, and to verify, independently, hardware readiness for software integration. Complete LCE/STC tests will be performed to verify the readiness of the LCE and STC for test operations. This test will be performed with the following special test equipment:

1) Spacecraft simulator;
2) LCE interface simulator;
3) Capsule STC interface simulator;
4) General- and special-purpose test equipment.



CONTENTS

4.0 FLIGHT SPACECRAFT SUBSYSTEM OSE

The Flight Spacecraft Subsystem Operational Support Equipment (OSE) provides test and checkout capabilities for the 12 individual subsystems of the spacecraft. It supports the subsystem type approval testing (TAT) and flight acceptance testing (FAT) and consists of individual subsystem test equipments that power and stimulate subsystems, measure responses, and evaluate functional performance. By providing a functional interface between the spacecraft subsystems and the central data and control system (CDCS) during prelaunch operations, this SS OSE supports testing of the assembled Flight Spacecraft and Planetary Vehicle; part of this same SS OSE also supports checkout of the Planetary Vehicle in launch countdown operations.

4.1 GENERAL

This section describes the general design features of the OSE for each of the 12 spacecraft subsystems listed in Table 3.1-1. There are 90 racks of SS OSE, of which 75 are used in the STC data reduction and display area, along with one control console in that area.

4.1.1 Summary

Typically, each Flight Spacecraft SS OSE consists of a test program control unit, a response evaluation unit and a stimuli generation unit. Detailed functions of each SS OSE are described in Sections 4.2, 4.3, and 4.4.

General characteristics applicable to SS OSE are summarized below.

The test routine for the SS OSE is assembled into a punched-tape program and manually inserted into the tape reader of the test program control unit. On demand, the tape reader reads out the program, which consists of the test routine, the stimulus selection for each input circuit, subsystem turn-on and turn-off, sequencing for application or removal of power where otherwise sudden on-off power transients might damage the subsystem, and an emergency subsystem power-off mode that can be initiated at any time during the test routine.

The Flight Spacecraft Subsystem OSE are of a "fail safe" design. The design goal has been to eliminate all failure modes which could induce a failure in the subsystem under test. The SS OSE design, however, incorporates automatic detection and emergency power shutdown features for protection against any critical condition which might originate within the subsystem under test.

To make manual adjustments on the subsystem under test, the test operator can initiate a hold, or stop, at any point in the test routine. He also can advance one test step at a time or skip portions of the routine. The hold, step, and skip functions are part of the manual operating mode.

The manual operating mode primarily supports the subsystem engineering and TAT test phases. The SS OSE evaluation unit can be programmed manually to monitor several output parameters. Hard-copy print-outs can be programmed to print subsystem status at predetermined intervals. From the control unit, all the SS OSE stimulus channels and signal measuring devices can be selected for performing required tests.

The SS OSE displays subsystem response signals for visual evaluation, the visual data being presented as a decimal display, using such instruments as digital voltmeters and counters. Section 4.2 describes details of certain other special display devices used. In addition, each SS OSE is provided with a magnetic-tape recorder to record subsystem response signals in IBM digital format, providing a rapid and efficient means for compiling subsystem trend data.

During the STC operation, the SS OSE is used as a functional interface between the spacecraft and the STC. For this, each OSE is provided with an STC interface connector that provides the subsystem test adapters with access to status and test point data on the SS OSE and the spacecraft. The STC interface connector provides circuits for direct access to raw test data (stimuli and responses) from the spacecraft test circuitry, for selection of subsystem outputs and their connections to the terminating load impedance and the output-signal-measuring device, and for selecting the upper and lower limits for each output response signal.

In addition, the tape program contains an SS OSE self-check sequence at each end of the test routine. Before a subsystem test can be initiated, the tape reader must step through the first self-check sequence and, before completion, through the second; this verifies the integrity of the OSE test equipment. A limit violation during any part of a test routine causes the tape reader to return or advance to the nearest self-check sequence and perform a recheck, after which the tape reader returns and stops where the limit violation was found; it remains there until the fault is cleared or a manual advance is initiated. During self-check return cycles, further subsystem stimulation is blocked until the check is complete. If the self-check reveals an OSE fault, interlock circuity is actuated to prevent further subsystem testing. When the OSE fault has been cleared, the interlock gates are reset manually and the complete subsystem test routine is reinitiated.

The subsystem being tested receives both a.c. and d.c. power from the OSE, actuated through a power-up sequence at the beginning of the tape program and removed by a power-down sequence at the end. These sequences are needed for control in directing application, sequence, and selection of appropriate system levels and stimuli; in obtaining subsystem OSE status; in interlocking to prevent inadvertent application of power and/or control signals; in interlocking subsystem OSE shutdown; in directing the SS OSE automatically from the central computer located in the STC; and in obtaining alarm or critical status data.

The spacecraft propulsion subsystem requires supporting equipment that verifies and conditions the subsystem for flight. Equipment required provides for propellant and solid-motor loadings, pressurization system loading, and verification of spacecraft subsystem status and performance. This SS OSE contains equipment required for testing, servicing, and handling the spacecraft midcourse-correction and orbit-insertion propulsion systems.

Although presently planned as government-furnished equipment (GFE), the Science Subsystem OSE plays a key role in the success of the Voyager missions, warranting an explanation of its structure and functions. It is used to validate the proper functioning of each Science Subsystem experiment in its operational environment. This OSE furnishes trend data for failure prediction and provides facility for malfunction isolation. It is used for subsystem testing, subsystem verification testing, subsystem integration testing, Seattle system testing, Kennedy Space Center (KSC) system testing, countdown and launch testing, and mission testing. Each science Subsystem test set contains a record of the current status of each instrument in the subsystem.

4.1.2 Applicable Documents

In designing the Flight Spacecraft SS OSE, extensive use has been made of the specification and guideline documents listed in Section 2.1. In addition, to form a clear understanding of the interfaces with, and the function of, the Science Subsystem OSE, reference has been made to the following documents.

1) EPD-250, "Mariner Mars 1969 Orbiter Technical Feasibility Study," NASA/JPL, 16 November 1964.
2) OSE/MC-4-210A, "Functional Specification Mariner C, Operational Support Equipment Science Subsystem," JPL, 2 October 1964.

4.1.3 Design Constraints and Requirements

Design of the spacecraft SS OSE has been guided by instructions and guidelines contained in pertinent documents listed under Sections 2.1 and 4.1.2. Without exception, the design presented here adheres to the specific requirements and constraints listed, and has been augmented by Boeing experience.

Spacecraft Bus SS OSE--This OSE performs a complete, isolated test of each Spacecraft Bus subsystem. It provides complete testing of each subsystem, manual control of each subsystem to any operating condition and in any sequence, rapid and repeatable performance of any extensive subsystem test routine, application of subsystem power equivalent to that normally supplied by the Flight Spacecraft power subsystem, and variation of spacecraft parameters or externally supplied signals for performance and margin testing. The OSE monitoring input impedance limits loading effects on a Flight Spacecraft function to less than 1 percent of the measured function. OSE current limiting is provided on all high-energy interfaces with the spacecraft subsystems or with other OSE. The system is conservatively designed with mechanization to protect the spacecraft against any possible OSE failure. Environmental alarm monitoring is provided while the spacecraft or any of its subsystems are under environmental tests. Automatic self-check is provided without requiring interruptions of spacecraft operation. Fault isolation is possible down to the subassembly replacement level. Manual control and visual monitoring is provided of all Flight Spacecraft SS OSE interfaces, and monitoring and recording is provided for all functions of the subsystem test circuitry and all simulated interface functions. Interfaces with connecting subsystems are simulated to present signals and loads required to perform independent subsystem tests.

Propulsion Subsystem OSE--The Voyager Propulsion Subsystem OSE is similar to that of other unmanned spacecraft programs including Mariner, Surveyor, and Lunar Orbiter. Basic differences between support equipment requirements of Voyager and Surveyor or Mariner OSE are due to the greater propellant requirements for Voyager and the multiple monopropellant engines for Voyager.

The basic difference between support equipment requirements for the Voyager liquid midcourse and orbit-trim propulsion and Lunar Orbiter is due to monopropellant usage on Voyager versus bipropellants used for Lunar Orbiter.

Design limitations imposed on this OSE are in the areas of: handling, storing, modes of testing, and installation and testing of potentially hazardous components. These hazardous components include the solid-propellant motor, pyrotechnic squibs, toxic and corrosive hydrazine propellant, and high-pressure vessels.

The propulsion system test unit provides all testing functions for the propulsion system that are necessary to ensure propulsion system operational integrity. It tests the regulators and all squib valves and solenoid valves. It actuates the engine thrust-vector control (TVC) mechanisms and indicates the position of the vanes. It senses the nitrogen tank pressure and the pressures of the freon tanks, providing temperature-corrected comparisons with the required pressures. It tests all check valves and the propellant tank diaphragm. The unit monitors propellant leakage, propulsion system propellant and freon temperatures, and the ambient temperature. It also monitors pressures of the pressurization system, propellant tanks, and propellant feed system, and indicates and controls the firing circuit "safe" condition. The unit provides hardline command capability until lift-off, for the relief valve in the high-pressure portion of the pressurization subsystem, and it monitors the propulsion system during servicing and/or testing.

Science Subsystem OSE--Requirements for design of this OSE are being considered by the manufacturer of this GFE item.

4.1.4 Reliability and Safety

Components will be selected for long-life operation and will be derated to allow extended-life operation in the normal operation environment. Components must meet storage or operating environmental requirements of JPL Specification 8900. Wherever possible, components will be selected from the DSIF preferred-components list, JPL Specification 8905. Equipment design must conform to the environmental requirements specified in JPL Specification 8900. Unless otherwise specified, to perform its intended function, equipment must be capable of stabilizing within 1 hour after turn on. Equipment requiring stabilization times longer than 10 minutes must be operable in a stand-by mode. Equipment must be capable of being subjected to a test cycle, to validate performance for at least 50 percent of that equipment, within 10 minutes after turn-on from a cold start. Commercial equipment used must be capable of meeting the quality requirements outlined in JPL Specification 8907A for DSIF equipment.

Personnel and equipment safety considerations of general application to SS OSE are as follows: Personnel safety covers and shields are used to prevent inadvertent personnel contact with high voltage leads; sources of heat are insulated to prevent accidental burning of personnel; and standard safety factors, such as rounded edges, minimal protrusions, and avoidance of overhang are maintained throughout the design.

Equipment protective devices have been incorporated in this OSE to prevent cascading of failures to interfacing equipment. Transient and overvoltage monitors detect and isolate any voltage harmful to the subsystem under test and to its connecting OSE, and isolation buffers protect the subsystem circuits from SS OSE failures.

4.1.5 Design, Development, and Test

The detailed design of the spacecraft SS OSE will be carried out in a manner which, with highest reliability, can ensure fulfillment of the stringent quality specifications necessary for Voyager success. To be sure that this objective is met, clear design direction is supplied under a carefully formulated design, development, and test plan. The concept design is initiated and controlled by detailed specifics of the general design criteria outlined in Sections 4.1.3 and 4.1.4. Preliminary achievement of the basic design goals, within those design constraints, leads to refinement in the development process, culminating in final operational testing of complete subsystems. Final tests are the ultimate steps in a long succession of test, redesign, and retest cycles that begin with the parts selection process.

Off-the-shelf proven components will be used where possible; rather than developing new components, existing ones will be purchased. Selection of those for final purchase will be guided by documented evidence of Boeing tests conducted to determine that these components will not only perform their required functions but also will do so over wide operating ranges specified for each application. New design of functional elements will be undertaken only as a last resort, and these will be proven by thorough testing of breadboard prototypes in which standard parts and modules are used at derated levels. Packaged engineering test models (ETM's) will be built, tested, and adjusted to ensure adequate tolerance margins on operation of the composite assembly. Tests and margin prove-outs will be documented and assembled for thorough reliability checks.

Component, subassembly, and assembly development and testing will be followed by rigorous integration checks to ensure that complete subsystems and whole OSE systems function according to OSE operational requirements under the full range of environment variations. The integration will include the design of signal and facilities distribution arrangements and proveout of such designs through construction, rack-up, and wire bundling in final operational form. These final-form prototypes will be tested, evaluated, and reworked where necessary, again ensuring the fulfillment of specified operational design margins.

4.2 SPACECRAFT BUS SS OSE

This OSE consists of 10 separate subsets of OSE, one for each of the 10 major subsystems of the Spacecraft Bus. These SS OSE's are listed below, and each is discussed under its own heading in subsequent subsections.

1) Power;
2) Guidance and Control;
3) Data Storage;
4) Radio;
5) Telemetry;
6) Command;
7) Computer and Sequencer;
8) Structures and Mechanisms;
9) Pyrotechnics;
10) Temperature Control.

4.2.1 Power Subsystem OSE

The power subsystem OSE includes all of the equipment required to test the complete power subsystem, as well as each of the major power subsystem components down to the replaceable assembly or subassembly level. It provides control, monitoring, and data recording for testing the power subsystem when installed in the spacecraft. It can simulate essential portions of the power subsystem for realistic testing of replaceable-level power subsystem components.

The preferred power subsystem OSE design is basically the Mariner OSE design, modified as required by the differences between the Mariner and Voyager spacecraft designs. An important addition to the basic Mariner OSE concept is tape programming of test functions and tolerance limits.

4.2.1.1 Functional Description

The power subsystem OSE includes equipment required for subsystem bench test through system testing, prelaunch, and launch operations. The equipment is described in the following order:

1) Subsystem OSE used in the STC;
2) Subsystem OSE used for TAT and FAT;
3) Solar panel test set;
4) Battery bench-test set.

The OSE for TAT and FAT consists of the three racks of STC OSE, plus a test adapter rack, as shown in Figure 4.2-1. The solar panel test set consists of two racks and an exciter. The battery bench-test set is in one rack. The relationship among power subsystem OSE functions is shown in Figure 4.2-2.

Systems Test Complex OSE--The equipment used in the STC includes that for power and control, data acquisition, and data recording. The power and control console supplies external or simulated solar-array power and simulated battery power to the subsystems under test. The control provisions include control of power supplied to the spacecraft, tolerance detectors and alarms, and panel instruments for monitoring battery terminal and cell voltages, simulated solar panel voltages and current, external power voltage and current, d.c. regulator output voltage and current, and running time of both spacecraft and OSE. All commands can be generated manually or by program for the power subsystem in the spacecraft.

The "external power supply" provides regulated ground power or simulated battery power to the power subsystem in the spacecraft. External voltage

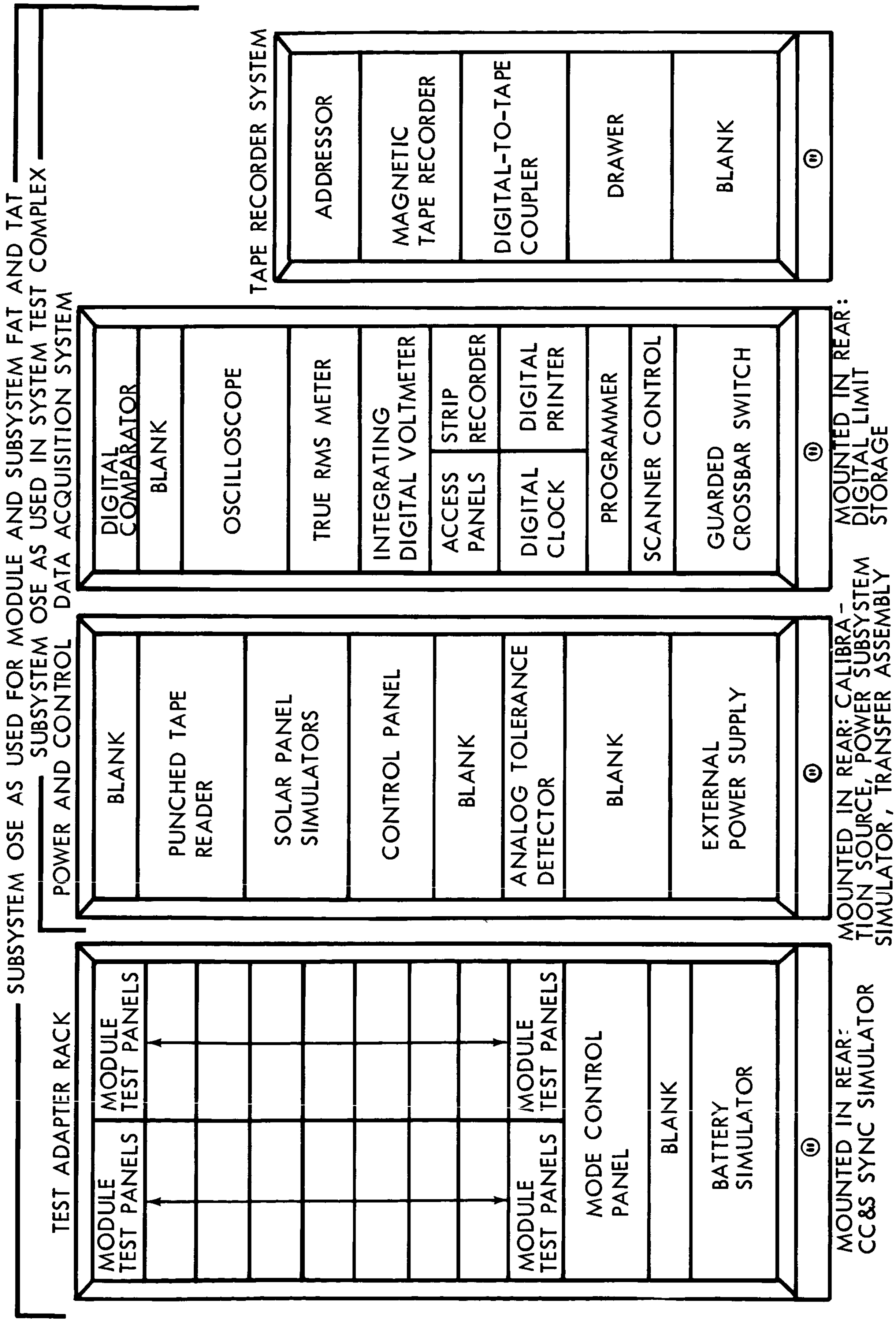


Figure 4.2-1: Power Subsystem OSE

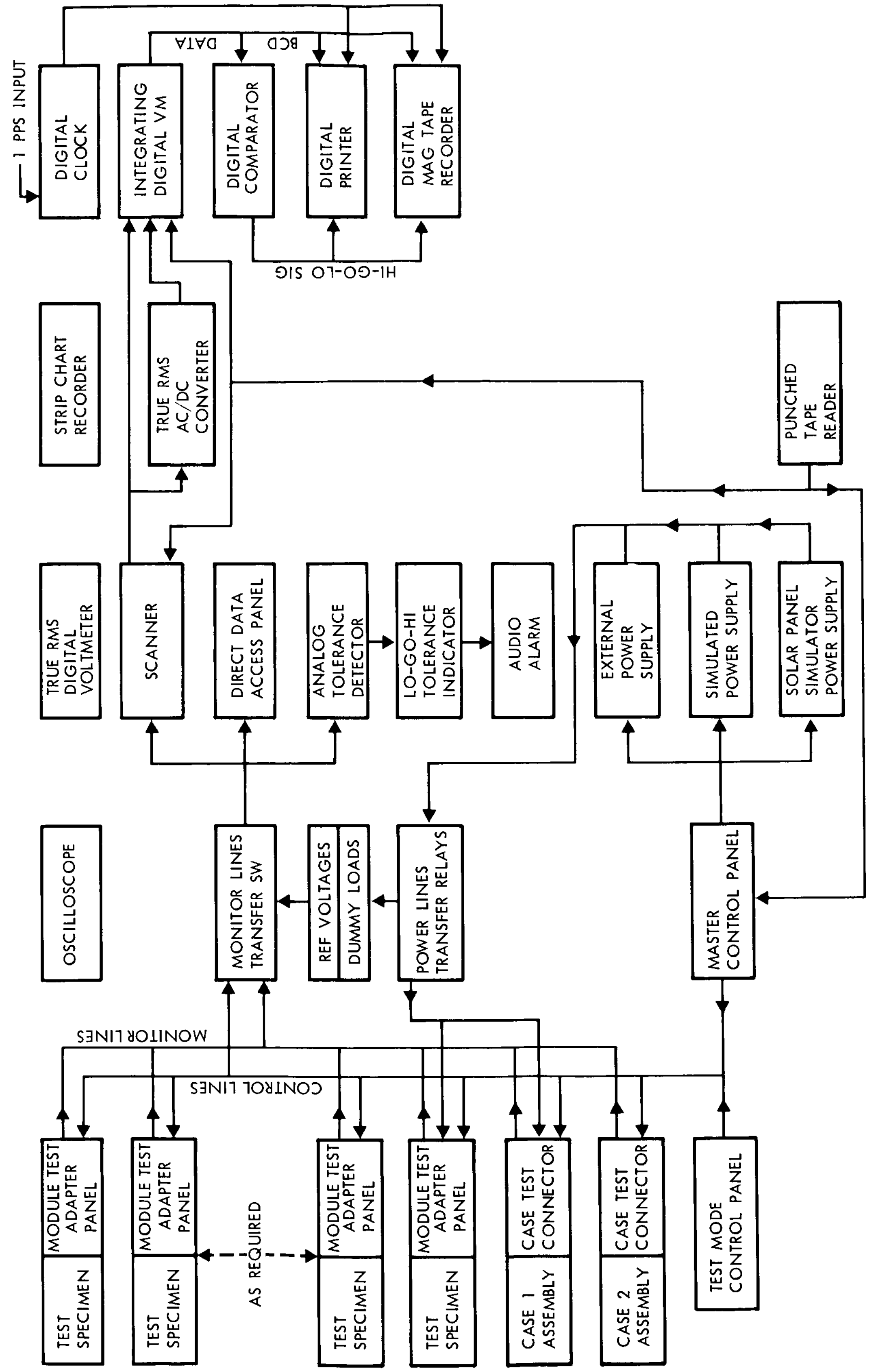


Figure 4.2-2: Power Spacecraft System OSE

sensing, current limiting, and voltage limiting are incorporated into the design. Voltage is programmable from zero through 100 volts d.c., and the current limit can be set anywhere between zero and 30 amps. Regulation from no load to full load is within 0.05 percent, and ripple less than 0.02 percent.

A dual solar-array simulator provides spacecraft power with characteristics closely matching the output of the solar array in space. The simulator is programmable through a voltage and power range equivalent to that encountered during an Earth-Mars flight. Open-circuit voltage can be varied from 0 to 100 volts. Maximum power points from 100 to 2500 watts are available.

The OSE contains self-test capability, provided by simulated power subsystem dummy loads and programmable reference voltages that simulate normal and out-of-tolerance conditions for all monitored circuits. The OSE can be switched to either the spacecraft power subsystem or to dummy loads and reference voltages. Power circuits and data access lines can be switched independently, thus permitting testing of all monitor functions without disturbing the normal operation of the power subsystem in the spacecraft. Self-test may be tape programmed or manually controlled.

The subsystem OSE has a 28-volt NiCad battery to supply critical panel indicators and control circuits during a facility power outage, thus maintaining control until normal power is restored, or allowing for a controlled shutdown.

Data Acquisition Console--The data acquisition console contains equipment for voltage, current, frequency, and temperature monitoring; digital recording on printed paper tape and magnetic tape; analog strip recording; and programmable digital comparison. Twelve full-time channels are provided to monitor voltage and current outputs of the batteries, regulators, and inverters. These channels are monitored continuously, both by analog tolerance detection in the OSE and by analog recorders in the central recording system. Two hundred other points in the power subsystem are sequentially monitored by a cross-bar scanning unit. The scanning speed depends on the mode of data recording, ranging from 5 points per second for the paper-tape recorder to 8 points per second for the magnetic-tape recorder.

An important component is an integrating average-reading digital voltmeter (Dymec) that virtually eliminates measurement errors caused by extraneous noise on the signal. The digital measurement is unaffected by grounds in the signal source, recorder, or programmer. The instrument input is floating and guarded, thus achieving effective common-mode noise rejection. The output from the voltmeter is a binary-coded decimal compatible with the digital comparator (Dymec), digital printer, and magnetic-tape coupler/recorder.

The incoming data are scanned by a guarded crossbar switch that maintains a high rejection of common-mode noise. The scanner accommodates up to 200 three-wire inputs. A wide-band Tektronix oscilloscope with a Type Z differential input amplifier is provided for monitoring data lines for wave shape, level, ripple, and noise.

The power subsystem OSE can be programmed either locally from a punched-paper-tape reader or remotely by a central data and control system.

3-1

Outputs from spacecraft direct-access cables and OSE digital data are provided for interface with the STC/SSTS adapter. Monitoring and evaluation are provided by both analog and digital tolerance detectors. Analog tolerance detectors permit full-time monitoring of the power subsystem on a limited number of monitor lines. An out-of-tolerance condition is indicated by an audible alarm and a panel indicator designating the circuit involved. An out-of-tolerance condition also produces a command signal to the digital data acquisition unit to start a complete scan of all monitor lines. The analog tolerance detectors are adjustable for calibration and are accurate to ±0.1 percent. Repeatability of indication is within 500 microvolts. Response time is 100 milliseconds maximum.

The digital output from the integrating digital voltmeter is compared with digital limits stored on a punched-tape program. This comparison is accomplished within 2 milliseconds by the Dymec digital comparator. Out-of-tolerance conditions are indicated by panel indicators and an audible alarm, and are flagged on both the digital printer and the magnetic-tape recorder.

TAT and FAT OSE--The OSE for the power subsystem flight-acceptance test (FAT) and type-approval test (TAT) consists of the three racks of STC OSE plus a test adapter rack. The test adapter rack provides a means of electrically connecting the power subsystem, either as individual modules or at a case level, to the power and control and data acquisition consoles. The rack also provides mechanical support for individual modules during testing, electrical loads, simulated battery power, substitute synchronizing signals, and forced-air cooling for power subsystem components while operating.

Solar Panel Test Set--The OSE required for performance, FAT, and TAT testing of solar panels consists of solar panel test console, Sun tracker, solar panel exciter, water bath, standard and secondary-reference solar cells, and solar panel deployment monitors. A typical arrangement of this OSE assembly is shown in Figure 4.2-3.

The solar-panel test console will be used during both the Sunlight and solar panel exciter tests to continuously load the panel from open-circuit to short-circuit conditions. The console is equipped to plot volt-ampere (I-V) curves and monitor solar-cell temperature and intensity of incident light. These data will be used to ensure that the panels are free of electrical defects and are capable of supplying the minimum required power under various conditions in space.

The solar tracker will be used during the solar tests to maintain the plane of the panel normal to the Sun's rays. Sun-tracking will be accomplished by Sun sensors and associated servomotors and controls.

The solar-panel exciter will be used to check the panel for electrical defects caused either in manufacturing or in transportation. The exciter will illuminate the panel with artificial light equivalent in intensity to that in space.

The water bath and the standard cells (secondary and reference) will be used in the solar tests. The standard cells maintained at a prescribed temperature by the water bath will measure the intensity of Sunlight in terms of equivalent-space-level intensity. Solar panel performance in space can be predicted by using the I-V characteristic, the observed panel short-circuit current, the equivalent-space-level intensity

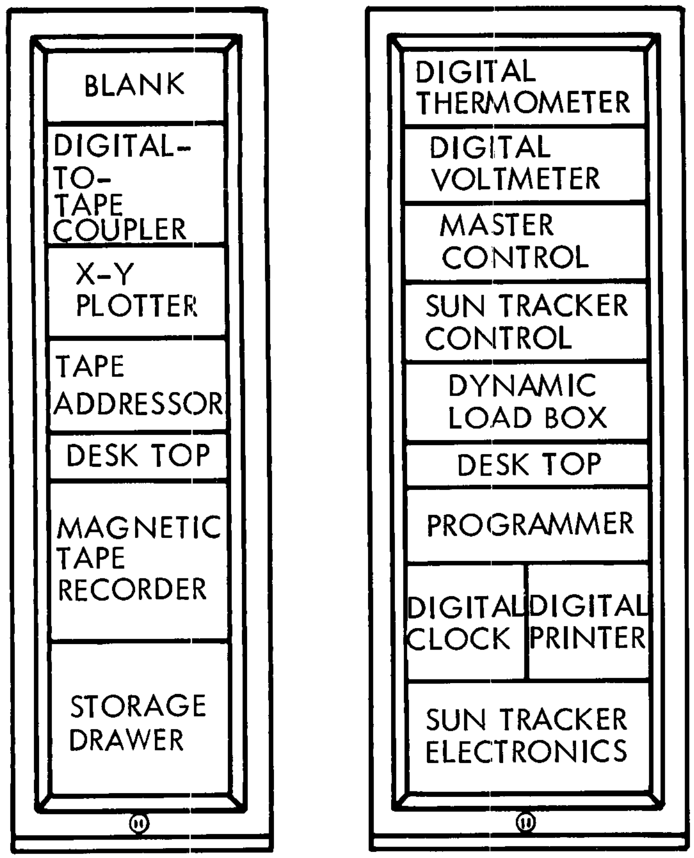


Figure 4.2-3: Solar Panel OSE

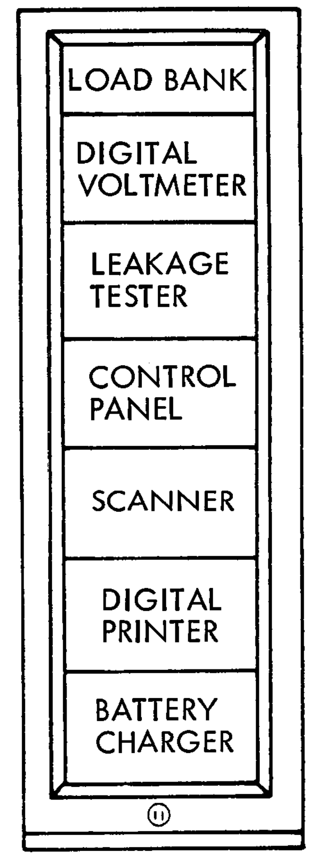


Figure 4.2-4: Battery Bench Test OSE

corresponding to this short-circuit current, and correcting the panel quantities to account for differences in intensity and temperature.

Temperatures will be measured to $\pm 0.5^{\circ}C$ and, when required, maintained to $\pm 1.0^{\circ}C$. Standard cell calibration will be accomplished by high-altitude balloon or airplane flights and will be accurate to ± 1.0 percent. The intensity of light will be measured to ± 1.0 percent and in the case of exciter will be maintained with a uniformity of ± 5 percent.

Solar-panel-deployment monitors will sense the initiation of deployment and latching of the panels.

Battery Bench Test Set--The OSE required for performance, FAT, and TAT testing batteries consists of a battery bench test OSE (Figure 4.2-4) and portable battery monitors.

The battery test bench consists of load bank, instrumentation, scanners, charger, and sequence programmer. This facility will be used to test each battery to ensure that the battery current, voltages, and ampere-hour capacity are within the specification. Individual cell voltage and temperature will be sequentially monitored during the test.

Accuracy of all measurement will be as high as present-day equipment and procedures permit. All electrical quantities will be measured by digital equipment accurate to ± 0.1 percent. Recorded data such as I-V characteristics will be accurate to ± 1.0 percent.

4.2.1.2 Physical Characteristics

The OSE for the STC is packaged into two 6-foot rack cabinets and a single 4-foot portable rack cabinet that contains the magnetic-tape equipment.

A third 6-foot test adapter rack houses the additional equipment for adapting the STC OSE for module and subsystem FAT and TAT.

The solar-panel test console consists of two 5-foot rack cabinets, each with caster bases. The solar-panel exciter equipment requires an air-conditioned area approximately 15 by 18 feet for operation. The battery test console consists of one 5-foot rack cabinet with caster base.

4.2.2 Guidance and Control Subsystem OSE

The guidance and control subsystem OSE provides for (1) acceptance testing of the guidance and control subsystem at both the subsystem and system level, and (2) performing the engineering evaluation tests required to verify design margins and environmental requirements. The subsystem test set is used at the subsystem level for manufacturing, TAT, and FAT testing and is later incorporated into the STC for system-level testing.

It provides accurate and repetitive testing in depth using precision stimuli and measurement equipment, a definite fixed configuration, and automatic control to permit extensive testing. SS OSE status can be determined at any time by self-check.

4.2.2.1 Functional Description

The guidance and control SS OSE functional block diagram is shown in Figure 4.2-5.

The test set verifies the guidance and control subsystem in dynamic operation by controlled stimulation of optical sensors, by the inertial reference unit, by simulation of switching commands, by measurement of

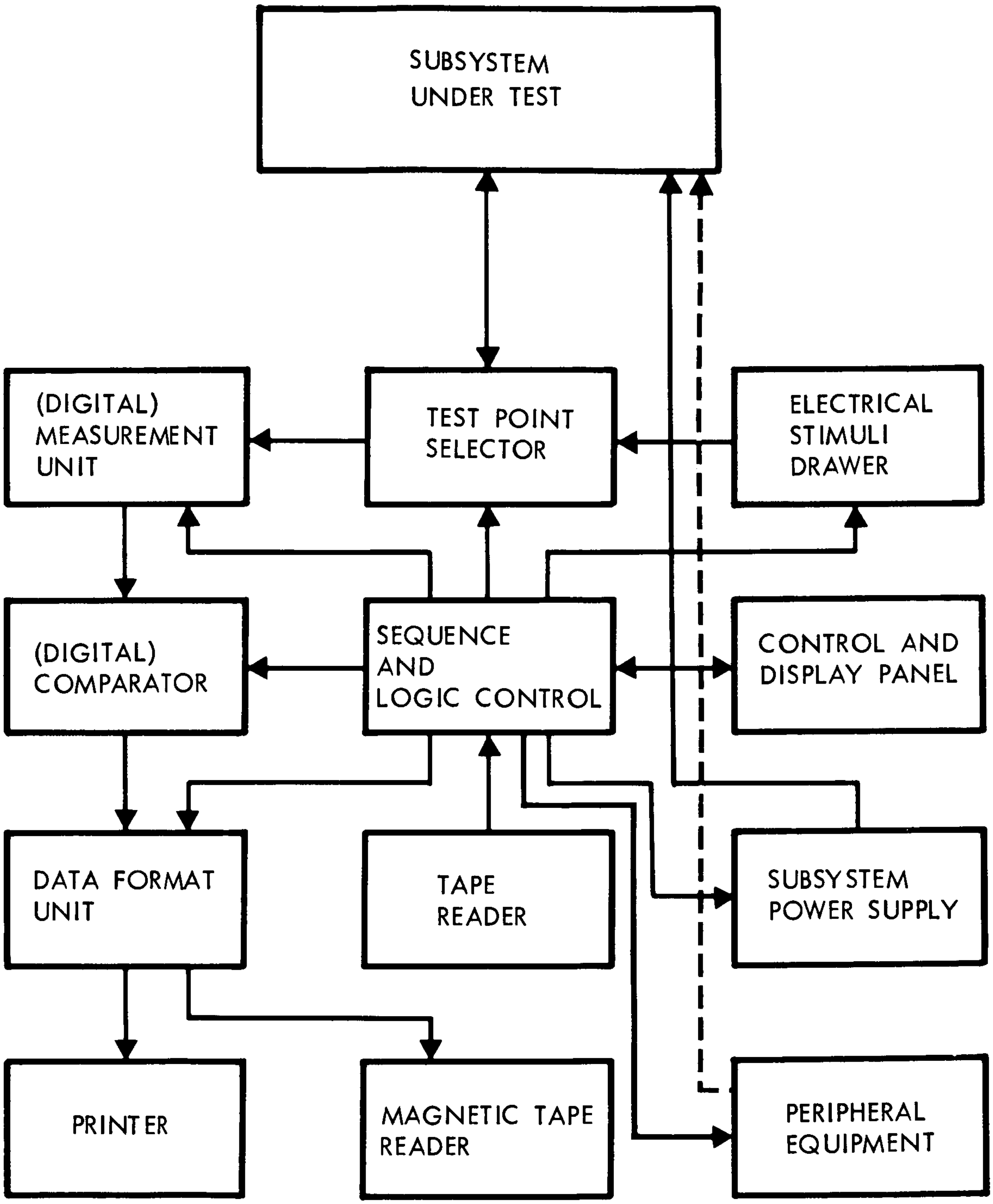


Figure 4.2-5: Guidance And Control Test Set

response to these inputs, and by comparisons of measured values with acceptable norms. Pertinent test data are recorded for future use in trend analysis.

In verifying the guidance and control dynamic operation, the test set performs the following functions.

1) Control--Selects the test point to be measured, provides excitation, sets up type of measurement and range, provides limits for evaluation, formats data for recording, and provides timing and commands. Control is provided by punched tape or manually by selection of test points, stimuli, measurements, and evaluation limits.

2) Circuit Switching--Provides an electrical, optical, or mechanical input; routes output to be measured; routes measurement to comparator and data format unit; and routes voltages to control points.

3) Simulation and Stimulation--Commands are simulated by discrete voltages to subsystem control points. Optical signals are provided for optical sensors and electrical simulation signals are provided for autopilot inputs.

4) Measurement--Electrical test point outputs are measured accurately with repeatability. Measurements include a.c. and d.c. voltages, frequency and elapsed time.

5) Comparison--The test set compares the digitized measurement values against limits and provides a "go" or "no-go" evaluation.

6) Format Data--Test data, including measured value, test number, or test conditions, and time, are put into an IBM format compatible with STC CDCS.

7) Record--Formatted test data are recorded on magnetic tape for use in trend analysis.

The test program for the attitude reference simulates the mission profile. A test of the gyro's capability to cage under high rates simulates the stabilization of the spacecraft after separation. The subsystem is mounted on the rate table, a tape installed, and the appropriate controls set up. After self-test is completed, tape commands set up a precision voltage for transmission to the rate table. The table rotates at a corresponding rate. Power is then applied to the subsystem; response test points are selected and switched into the measurement unit. Gyro rate is digitally compared against tape-programmed limits and results are displayed. Simulated C&S commands are provided by the stimuli drawer. The autopilot's control of reaction-control-system (RCS) jets, in response to gyro outputs, is tested by moving the rate table through a programmed angle and monitoring jet-vane-driver outputs. The rate table is controlled by commanding a low rate of rotation and comparing the output of the table position digital encoder with a programmed limit. The gyro position is also measured.

The guidance and control test set consists of an electronic test rack, rate table; star simulator; Sun simulator; planet, limb, and terminator simulator; an angle measurement adapter; a cable set; and program tapes.

The electronic test rack contains electronic assemblies required for the programmer evaluation, data format, display, and recording functions. It includes the electronic stimuli and subsystem power supply. The electronic test rack is shown in Figure 4.2-6. Programmer evaluation functions are performed by the following electronic test rack subassemblies:

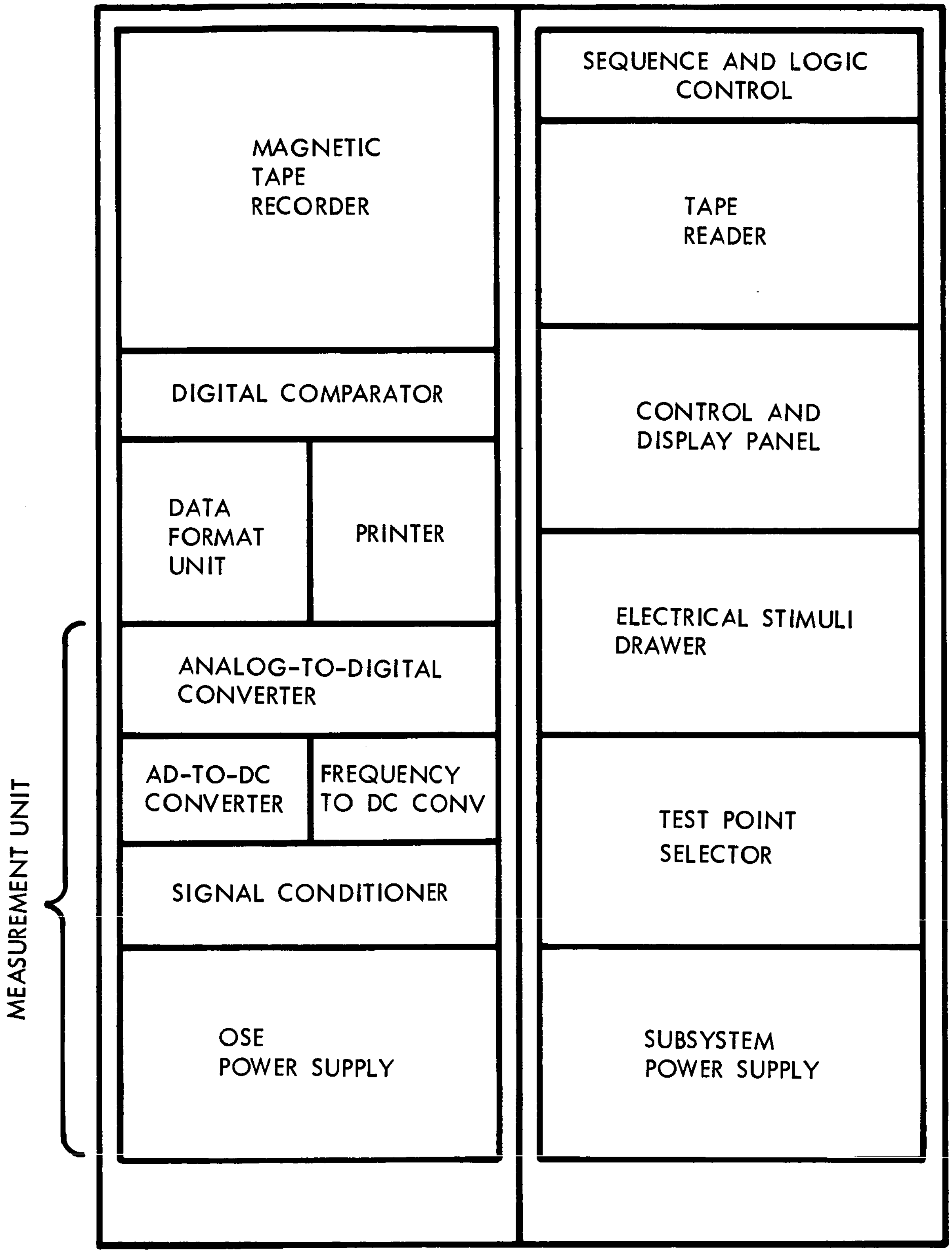


Figure 4.2-6: Guidance and Control Subsystem OSE

1) Tape Reader--An Engineering Electronic Company Photoblock tape reader Model PR2500-2 and the circuits to control and drive mechanism and condition and gate the outputs.
2) The Sequence and Logic Control--Receives test command data from the tape reader. Test commands are decoded and routed to the appropriate functional block for execution. Test data such as test limits, test value, and identification number, are routed to the comparator and data format unit. The sequencing and control logic verifies that test sequences are completed, advances the program to the next sequence, and performs self-check.
3) Test Point Selector--Provides input and output switching required to perform the various test services. Response inputs are switched to the measurement unit for measurement and evaluation. Stimuli are switched from the stimuli generator to the subsystem. The selector uses a crossbar scanner (Cunningham Model ST-2) for input test-point selection. Stimuli switching requires high current capability and uses reed relays.
4) Measurement Unit--Measurement functions require a digital voltmeter, a.c. converter, counter, digital register, and control circuitry. A Hewlett Packard integrating DVM, Model DY-2401 with an a.c. converter Model DY-2401B, provides voltage and frequency measurements.
5) Comparator--Hewlett Packard digital comparator, Model DY-2539A. Both display and electrical indications of HI/LO results are provided.

Simulation and stimulation functions are provided by subassemblies of the electronic test rack, and the stimulators, as follows:

1) Electrical Stimuli Drawer--Provides electrical input stimulation for subsystem functions. The stimuli consist of:

   a) Ramp voltages (2)-- 0 to ±15 volt (controllable in amplitude and time);

   b) Square wave generator--0 to 25 pps;

   c) D.C. scaler voltages, 0.01 volts to ± 15 vdc, accuracy of 0.1 percent;

   d) D.C. current-variable from 0 to 10 ma in 0.05-ma steps.

   Controls to select parameters, to initiate, to terminate, and to gate outputs are included.

2) Power Supply--Provides 100 volt, 2400-cps power and 19.2 kilocycle reference frequency to the test subsystem and monitors the voltage supplied to the subsystem. If an out-of-limit condition is detected, an alarm is initiated, power to the test subsystem is removed, and the test stopped. The output voltage can be adjusted for voltage-limit testing of the subsystem.

3) Star Simulator--An optical test fixture providing collimated light from a point source. The spectral content of the source approximates that of Canopus. Intensity is variable from 0.1 to 5 times that received by the Earth from Canopus to allow testing of upper and lower gating circuits and intensity of the readout signal. The axis of the simulated star and the pointing axis of the star sensor is established to 0.01 arc degree. The simulator has a mobility of ± 4 degrees about the nominal roll axis and a mobility of ±16 degrees about the nominal pitch axis. Simulator intensity is controlled from the electronic console.

4) Sun Simulator--A bright light source, a diffusion screen, and a collimating system. The axis of the simulator is aligned to the pointing axis of the Sun sensor within 0.01 degree. The simulator has a mobility of ± 6 degree about the pitch and yaw axes.

5) Planet, Limb, Terminator Simulator--A sphere mounted against a nonreflective background, illuminated by a single collimated light source. The spectral characteristics and intensity of the source and the reflectivity of the sphere approximate Mars light within the spectral response band of silicon and cadmium-sulfide photodetectors. The sphere and light are movable in azimuth and elevation for pointing response check and light phasing. Variations in Mars-spacecraft distance are simulated by (1) moving the sphere or (2) using various sized spheres. This simulator also serves to check the Earth sensor.

6) Tilt and Rate Table--A tilting rate table is used to test the attitude reference gyros and accelerometer. The table rotational rate is controlled from the test set at rates varying from 0.05 degree per second to 5.0 degrees per second. The table can:
   a) Be positioned with a repeatability of 30 seconds;
   b) Be tilted through an angle of 90 degrees;
   c) Accommodate packages weighing up to 100 pounds with dimensions of 24 by 24 by 24 inches. To obtain a near perfectly uniform rate, a table with a hydrodynamic bearing is used. The Davidson hydrodynamic balance (HDB) Model D500-103 provides these capabilities.

7) Angle Measurement Adapter--This device provides accurate measurement of the angle between the plane normal to the antenna pointing vector on the back side of the antenna paraboloid and the reference mounting surface plane for the antenna drive. The measurement mechanism is independent of the antenna drive and the regular shaft encoders of the antenna. Accuracy is determined to 1/2 the angular increment of antenna command angle. Any cross coupling is indicated.

Data format, display, and recording functions are accomplished in the following electronic test rack subassemblies:

1) Data Format Unit--Assembles test data and test identification data into IBM format compatible with the STC data processing system, and provides temporary storage registers; and the controls required to accept, readout, and synchronize the data. Identification data are entered from the tape program and/or the manual data entry on the control panel.

2) Control and Display Panel--Contains all the controls and displays needed for the test operator to initiate, override, and stop any test sequence; to evaluate test results; and to analyze and resolve operating problems. Test number, measured value, and test limits are displayed by digital readouts. HI/GO/LO, self-test, operating modes and alarms, and other subsystem/OSE status items are displayed on indicator lamps. Running-time meters record OSE operation time and subsystem power-on time. A meter movement with a selector switch is used to measure subsystem power and OSE voltages and currents. Selection and initiating switches are provided to control operating modes. Thumbwheel switches are used to insert the data,

test location, operator's identification, and other data into the data format memory. Specific tests can be called up through a similar set of switches. Circuit breakers control application of power to the test rack, rate table, and optical simulators. Application and removal of subsystem power is controllable by tape program with manual backup. The critical status monitoring and emergency shutdown circuits, contained in this subassembly, are functionally and electrically isolated from other functions. Failures in tape reader, the sequence and logic control, the input selector, or other subassemblies will not affect its operation. Separate power supplies are provided.

3) Data Recording--All test data are recorded in IBM format ready for use in trend analysis without conversion. The test number, test time, and measured values are recorded by an incremental magnetic recorder, Model 1400. A Hewlett Packard digital recorder, Model 562A, provides hard-copy data for on-the-spot evaluation and data package records.

4.2.2.2 Interfaces

Interfaces are given in Table 4.2-1.

4.2.2.3 Performance Parameters

Performance parameters are given in Table 4.2-2.

4.2.2.4 Physical Characteristics

Two standard JPL racks house the guidance and control subsystem OSE electronics. Pullout drawers allow quick replacement. Displays and controls, grouped together in one location, are situated at a convenient

Table 4.2-1 Interfaces

Type of Interfaces	Description	Interfacing Element	Number of Interfaces
Electrical	Discrete command simulation subsystem inputs from C&S and command	From attitude reference and autopilot to electrical test rack.	105 signals
Electrical	Response from attitude reference and autopilot in analog and discrete signal includes telemetry.	From attitude reference and autopilot to electrical test rack.	105 signals
Electrical	Analog stimulation signals to propulsion system	Electrical test rack to attitude reference and autopilot	26 signals
Electrical	Analog signal simulating actual feed-back	From electrical test rack to attitude reference and autopilot	29 signals
Electrical	Primary power 50-volt peak square wave 2400-cps and 9.2-kc clock signals	From electrical test rack to the attitude reference and autopilot	10 signals
Electrical	Control of rate table power, rotational rate, and position	From the test rack to the rate table	24 signals
Mechanical	Support and position attitude reference and autopilot package	From the rate table to the attitude reference and autopilot	
Electrical	Control power density and position of an optic simulation for star simulation, sun simulation, planet simulation, and limb/terminator simulation.	From electrical test rack to optical simulation	6 per simulation
Optical	Control alignment and intensity of optical simulators with respect to optical sensors	From the optical simulators to optical sensor	
Primary Power	120 v. a.c., 60 cycles, single phase	From facility power system to the electrical test rack	3 signals

Table 4.2-2 Performance Parameters

Parameter	Verify
AUTOPILOT	
Jet vane and secondary injection valve rate and position gains for insertion, midcourse-correction, and orbit-insertion maneuvers	Autopilot gains are verified by commanding autopilot into the mode of interest and providing a known input value. The output across the simulated load is measured and compared with precomputed values.
Limit-cycle operations threshold levels and minimum valve actuation time	Threshold and dead band are determined by applying an input voltage just under the specified threshold and monitoring for an output. The input is increased by increments until the output switch fires. The input voltage is decreased until the switch turns off. Correct response under all modes of operation is verified. An input voltage well above the threshold is applied for 10 milliseconds. The driver output into the simulation RCS valve is verified to be 20 milliseconds.
Gyro, mass torquing rate	The attitude reference is rotated at 4000 degrees per hour and gyro input is monitored to verify that the gyro is caged. Rotational rate is decreased to 1000 degrees per hour and gyro range switching operation verified.
Drift, drift bias, and drift sensitivity	Attitude reference is mounted in various positions on the rate table and rotated to determine various drift components. Both access outputs are monitored.
Temperature sensitivity	The ambient temperature is varied and drift temperature response is determined.
Torquer slew and linearity characteristics	Torquer inputs are commanded and gyro torquing rates are measured.
Accelerometer linearity and threshold	Attitude reference is subjected to various accelerations and outputs monitored. Threshold is verified to be 1 micro g.
Scale Factors and scale factor temperature sensitivity	Scale factor is determined from tests of several accelerations. Ambient temperature is varied to check sensitivity of 1 ppm/^{o}F.
Bias stability and bias temperature sensitivity	Zero input bias is determined and checked against previous history. Ambient temperature is varied to determine bias temperature sensitivity.

Table 4.2-2 (Continued)

Parameter	Verify
Canopus Sensor	
Tracking accuracy	Sensor output is measured at off-axis roll angles. The zero output position must be within 0.1 degree of the mechanical axis as determined by the mounting provision.
Field of view	Output is measured to determine that the field of view is ±2 degrees in roll. The output of the field of view is checked for correct response. Pitch gimbaling is checked for operation in 4.6 degree steps.
Time constant	Response to a stepped input and star magnitude is measured to verify that the time constant is 0.5 second.
Sun Sensor -- Fine	
Tracking accuracy field of view and scale factor	Sensor output is measured over entire field of view. Zero output position must be within 0.15 degree of the mechanical alignment axis. Linearity of the field of view to ±5 degrees is verified and total output field of view is determined. The scale factor is determined to be 1 volt per degree.
Time Constant	Response to step input in Sun magnitude impact against 10-millisecond requirements.
Sun Sensor - Coarse	
Tracking accuracy and time constant	Sensor output is verified to be aligned within 1 degree and have 10-milliseconds response to the change of magnitude.
Planet Sensors	
Tracking accuracy	Alignment of sensor zero output is checked against instrument axis to be 0.1 degree (planet 0.1 degree, limb terminator 1 degree).
Time constant	Sensor response to the step input is verified to be 0.1 second (planet) and 1 second (limb terminator)
Altitude Range	Output sensor at a light intensity equivalent to a distance of 8 to 50 x 10^4 kilometers (planet) and 10^3 to 10^4 kilometers (limb terminator) is measured and compared against limits.

height. Peripheral equipment--rate table, Sun simulator; star simulator; planet, limb, terminator simulator; angle-measuring device--are connected and controlled by cabling connected at the back of the rack. A single 120-volt, 60-cps, single-phase drop provides test set power.

4.2.3 Data Storage Subsystem OSE

The data storage SS OSE checks out the Voyager data storage subsystem. The SS OSE performs two functional test operations: diagnostic and operational. The diagnostic test involves subsystem fault detection and isolation to a replaceable subassembly level. Operational tests execute proper mission sequences under environmental stress to certify the subsystem. Both the operational and diagnostic test operations are performed at the subsystem level during the manufacturing, TAT, and FAT phases. The SS OSE is also capable of performing portions of the spacecraft checkout operations at the STC and LCE locations.

4.2.3.1 Functional Description

The basic design of the SS OSE permits the operator to select either a manual, semiautomatic, or automatic test mode. The automatic mode eliminates the possibility of operator error associated with the repetitious testing of many signals.

The design also includes an SS OSE self-check sequence that can be activated in any of the above modes.

Sufficient SS OSE flexibility has been provided to support test growth and subsystem changes resulting from a change in mission assignment.

The SS OSE provides clock pulses and simulated data signals that drive the storage recorders. Simulated control signals for record, playback, on/off, and recorder speed control are sequenced by the SS OSE. In addition, it simulates command override for record and playback control, plus the master digital multiplexer, clock and data outputs. Playback sync signal, normally supplied by the master digital multiplexer, is also provided by the SS OSE.

Playback of simulation data, and subsystem response to mode, control, and command are monitored and measured by the SS OSE. Transducer outputs of engineering data are sampled, buffer stored, and displayed at required intervals.

The SS OSE powers the subsystem during testing. Protective measures such as subsystem dc power monitors and temperature sensors are incorporated to avoid damage to the subsystem.

The data storage SS OSE consists of the five major assemblies, interconnected as shown in Figure 4.2-7. The major assemblies and their operating sequences follow.

Signal Simulation and Stimulation Assembly--Simulation data are provided as serial PCM NRZ pulses. Data and sync rates are indicated in Volume A, Section 4.1.5. Data signals, degraded in voltage, test the expected gain margin of the record electronics. Pulse-width degradation is also provided. The 2400-cps, 50-volt test power is varied under program control by $\pm$ 15 percent in excess of specified voltage limits. Rise and fall time variation of the power square wave is provided. Noise exceeding specified voltage and duration is injected into the power to test the

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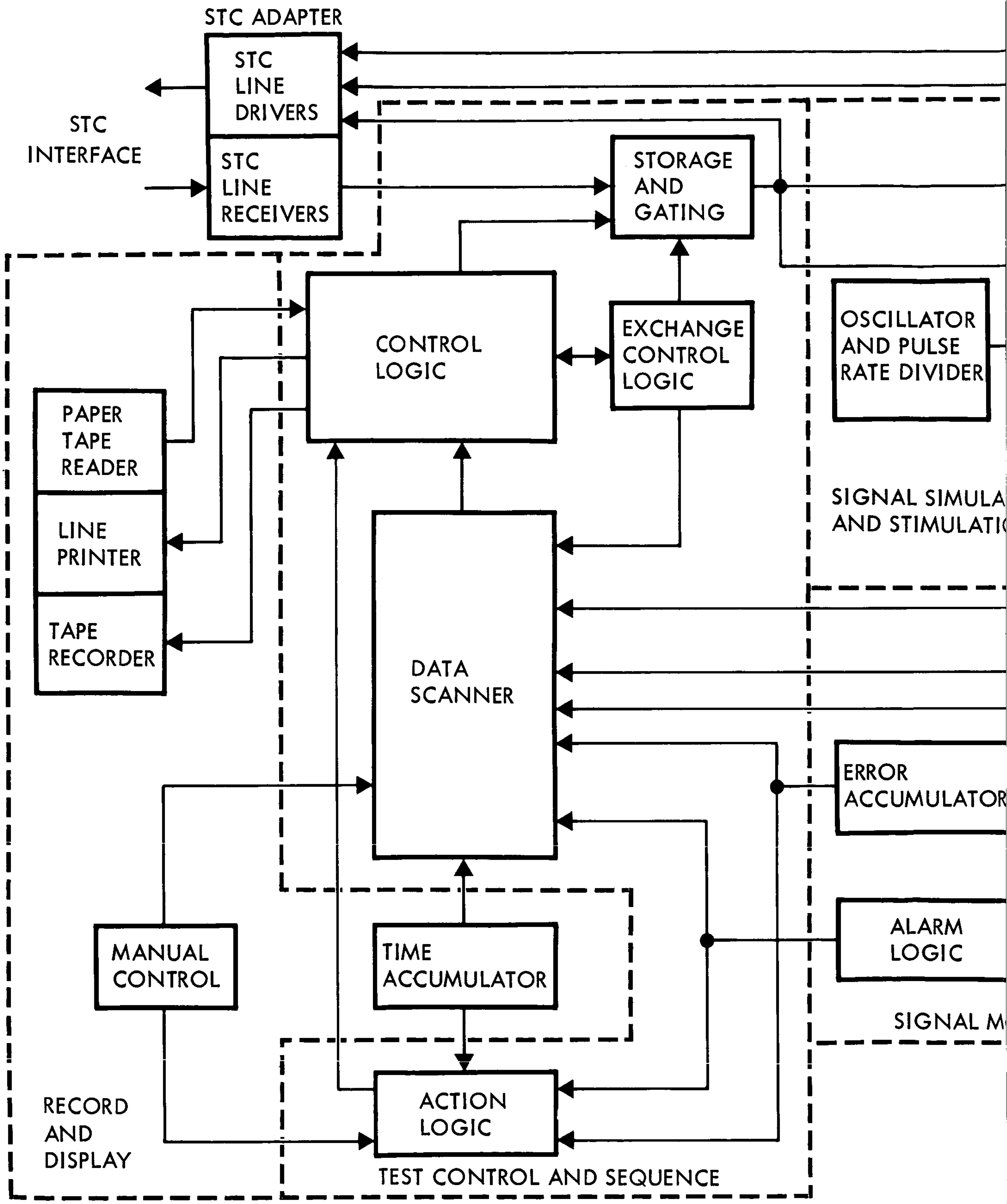

STC ADAPTER
STC LINE DRIVERS
STC LINE RECEIVERS
STC INTERFACE
STORAGE AND GATING
CONTROL LOGIC
EXCHANGE CONTROL LOGIC
OSCILLATOR AND PULSE RATE DIVIDER
PAPER TAPE READER
LINE PRINTER
TAPE RECORDER
SIGNAL SIMULA
AND STIMULATI
DATA SCANNER
ERROR ACCUMULATOR
MANUAL CONTROL
TIME ACCUMULATOR
ALARM LOGIC
SIGNAL M
ACTION LOGIC
RECORD AND DISPLAY
TEST CONTROL AND SEQUENCE

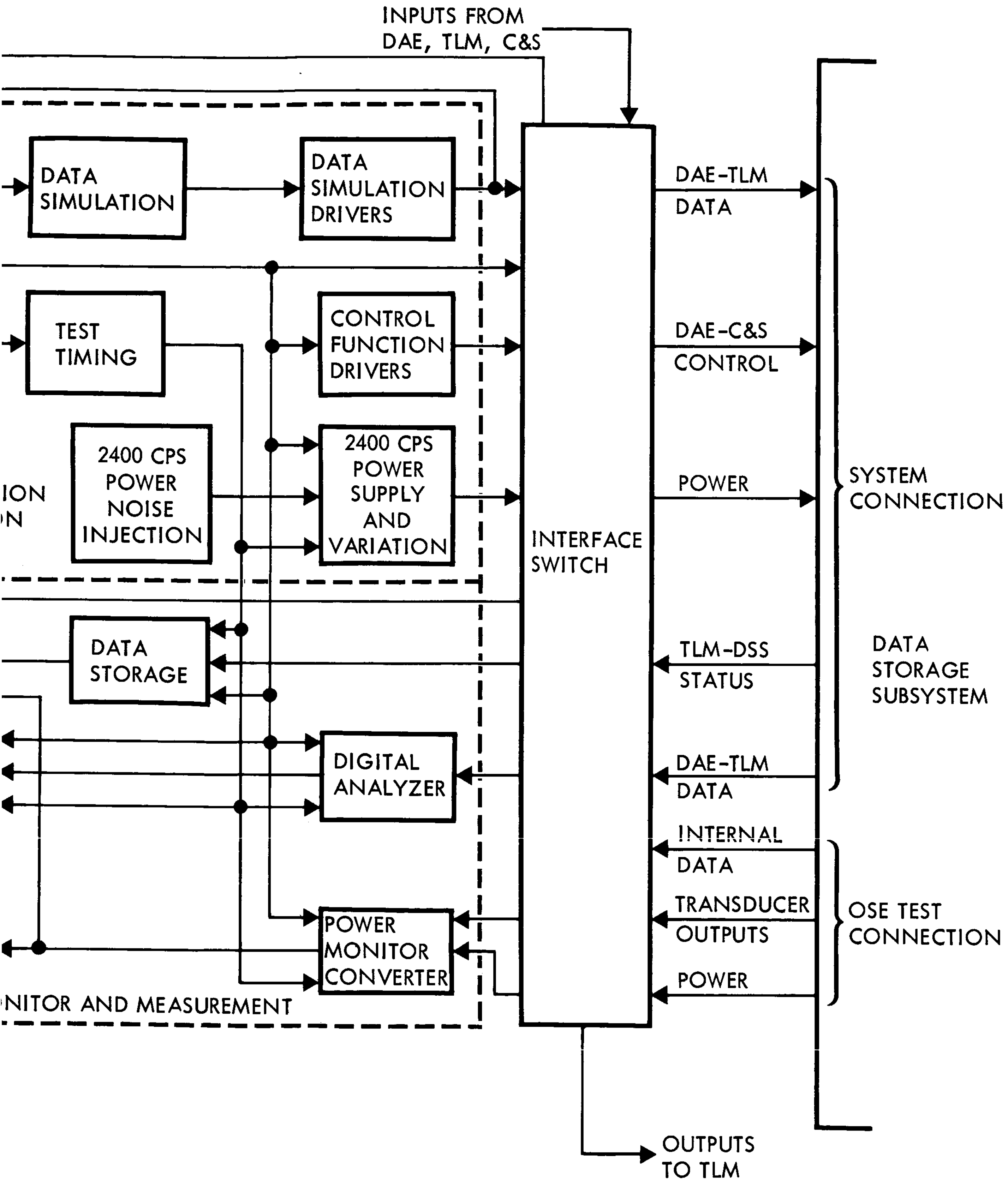


Figure 4.2-7: Data Storage Subsystem Test Equipment

subsystem susceptibility to noise. Current limiting in power interfaces and in any direct-coupled interfaces prevents internal subsystem failures from multiplying.

Signal Monitor and Measure Assembly--Data signals are monitored for degradation, bit error rate, bit rate, and skew. Pulse widths are measured and compared and pulse rates are counted. The 3-bit source identification code to the master digital multiplexer from the planetary science recorders is decoded. The 2-bit source code from the spacecraft data, and fields and particles recorders is decoded. Analog voltages such as subsystem dc power, and temperature and pressure sensor outputs, are converted by an analog-to-digital converter and recorded. Subsystem power voltages and temperature sensor outputs are compared with programmed limits by OSE alarm circuit. Should a temperature or power alarm condition be detected, power is disconnected from the subsystem by the interface switch.

Test Control and Sequence Assembly--A special-purpose digital device with a 4000-word programmable memory supplies test control and sequence. This unit also performs quantitative comparisons and formats both trend and display data. The memory is programmed by punched paper tape or manually.

Record and Display Assembly--Test data are displayed on a hard-copy line printer. The data are identified as to test number, specific information, units, date, and time of day. Magnetic tape is used for recording trend data to provide a direct interface to the trend-data processing equipment. Test data displayed consist of every measurement taken during a test.

The data required for trend analysis consist of measurements of internal subsystem voltage levels, bit error rate, jitter, skew, tape speed, temperature, pressure, input power, power factor, and phase-lock-loop error signal.

STC/SSTS Adapters--The function of the STC/SSTS adapter is described in Section 3.1.5.2.

4.2.3.2 Interfaces

The data storage SS OSE provides the subsystem interfaces listed in Table 4.2-3.

To support STC testing, an interface is provided to enable the STC to control the system test functions performed by the subsystem test set. STC access to raw test data is provided through the interface.

4.2.3.3 Performance Parameters

The data storage SS OSE performance parameters are determined by the subsystem input and output signal requirements. The majority of the SS OSE performance parameters have been generated to comply with the DAE subsystem input parameters to the data storage subsystem. They are as follows:

1) Simulated Fields and Particles data
    a) Data signal 825 bps
    b) Data clock signals, simulating DAE and master digital multiplexer clocks.
    c) Control signals from DAE

TABLE 4.2-3

ITEM NO.	INTERFACE DESCRIPTION	SIMULATED INTERFACING SUBSYSTEM
1	Provide 2400-cps power, two lines: prime, standby	Electrical Power
	Provide simulated digital data signals:	
2	80 bps and 288 bps and 60 bps	Telemetry
3	50,000 to 200,000 bps (capsule relay)	Radio
4	825, 4800, 57,600 bps	Data Automation
	Monitor simulated digital data signals:	
5	11.5, 59.4 or 15.4 bps	Telemetry
6	7200, or 1200 bps	Telemetry
	Provide clock signals:	
7	80 or 288 or 60 bps	Telemetry
8	7200, or 1200 bps	Telemetry
9	825, 4800, and 57,600 bps	Data Automation
	Provide tape recorder control signals:	
10	Science data recorders (5 recorders)	Data Automation
11	Record on-off; reproduce on-off	Command
12	Capsule relay recorder:	Data Automation
	Record on-off; reproduce on-off	Computer and sequencer
	Spacecraft data recorder (maneuver unit):	Computer and sequencer
	Record on-off; reproduce on-off	Command
		Data Automation
	Provide reproduce tape speed mode signals:	Computer and Sequencer
	Provide reproduce tape speed mode signals:	Command
	Monitor tape record status signals:	
	End of tape; start of tape; recording data; reproducing data; unit malfunction	Data Automation
	Reproduce status (recorder identification)	Telemetry
	Monitor clock signals for synchronization of reproduced data	Telemetry
	Monitor engineering data	Telemetry
	Recorder temperature (7 units)	
	Recorder pressure (7 units)	
	Monitor input to the record head drivers	OSE
	Monitor output from playback amplifiers	OSE
	Monitor internal subsystem dc power	OSE
	Monitor environmental temperature alarm sensor	OSE

2) Simulated IR and UV spectrometer data
   a) Data signal, 4800 bps
   b) Data clock signal from DAE
   c) Control signals from DAE
3) Simulated IR Scanner data
   a) Data signal, 4800 bps
   b) Data clock signal from DAE
   c) Control signals from DAE
4) Simulated Photoimagery data (Recorder 1)
   a) Data signal, 57.6 Kbps
   b) Data clock signal from DAE
   c) Control signals from DAE
5) Simulated Photoimagery data (Recorder 2)
   a) Data signal, 57.6.Kbps
   b) Data clock signal from DAE
   c) Control signals from DAE
6) Simulated Capsule Relay data
   a) Data signal, 50 to 200 Kbps
   b) Data clock signal from DAE
   c) Control signals from DAE
   d) Capsule relay data recorder control signals

The spacecraft systems also provide inputs to the spacecraft data recorder. These input parameters are also generated by the SS OSE and include the following:

1) Command subsystem data, consisting of record and playback control signals;

2) Computer and sequencer data, consisting of record and playback control signals;
3) DAE input, including (1) fields and particles data, 825 bps; and (2) fields and particles data clock signal;
4) Master digital multiplexer input, consisting of maneuver data signal and data clock signal.

The spacecraft supplies a C&S mode control signal that drives the DAE and the capsule relay data recorder. The SS OSE simulates both of these parameters.

The data storage OSE measures the following subsystem output parameters:

1) Output from the recorder playback identity generator;
2) Output from planetary data playback signal selector;
3) Fields and particles recorder identity signal;
4) Spacecraft data recorder identity signal;
5) Signal selector output
    a) Data signal,
    b) Clock signal.

    (for either spacecraft data recorder or fields and particles data recorder)

4.2.3.4 Physical Characteristics

The data storage OSE consists of three standard NASA/JPL racks, and a two-bay console. The console supports the subsystem test fixtures, and contains the operator position and control panel. Approximate weight of the OSE is 1500 pounds. Power consumption is approximately 1500 watts.

4.2.3.5 Trade Studies

Computer Versus Special-Purpose Equipment--Special-purpose logic equipment can perform the job of test sequencing and data comparison not only at a much lower cost than can a large-scale computer, but also at data rates more than adequate to satisfy the information exchanges required by DSE and OSE, making computer formatting and analysis unnecessary. Special-purpose logic equipment can operate well within required test times and can be put together in a form that actually enhances manual operation. Programmed by a punched tape, it can function automatically, and can produce a tape for trend-data analysis.

Automatic Testing--The only major advantage of conducting the subsystem test manually is its lower initial cost. Performance of the test, repeatable many times in identical sequences, is an extremely important requirement. Repetition requires fast performance. Only with automatic test equipment can both speed and reliability be achieved to the degree required.

4.2.4 Radio Subsystem OSE

The test concept for the Voyager spacecraft radio subsystem is similar to that employed for Mariner C. The general subsystem OSE configuration is also quite similar, except that this OSE includes the capability for testing spacecraft relay radio link equipment, a function not required of the Mariner C radio subsystem OSE.

4.2.4.1 Functional Description

The radio SS OSE provides the functional capabilities to:

1) Power the spacecraft radio during test procedures;

2) Control the spacecraft radio modes during test;
3) Control the test operations;
4) Monitor the radio subsystem operation;
5) Measure the radio subsystem performance;
6) Display signal parameters and status during test;
7) Record test and trend data from subsystem TAT and FAT;
8) Self-check the SSTS.

Power--The SS OSE provides (1) up to 100 watts of 2.4-kc square-wave power, variable in frequency and voltage up to 15 percent, and (2) up to 350 watts of unregulated dc power, variable from 37 to 100 volts.

Spacecraft Radio Mode Control--The SS OSE provides radio subsystem mode control by simulating (1) command signals from the C&S and command subsystems, and (2) automatic failure-switching functions within the subsystem (see Volume A, Section 4.1.6).

Test Operations Control--The SS OSE provides (1) STC control for system tests, including the selection of the test to be performed, the control of power application, and external signal application; and (2) local control for subsystem tests by the selection and control of input stimulus, measuring equipment, display device or devices, and recording equipment.

Monitoring Capability--The SS OSE provides instrumentation for monitoring overtemperature, overcurrent, and excessive VSWR.

Radio Subsystem Performance Measurement--The SS OSE performs all required STC system- and subsystem-level tests, as listed in Table 4.2-4. In addition, it provides fault-isolation capability to the following:

TABLE 4.2-4

RADIO SYSTEM LEVEL TESTS	S-Band Radio Equip-ment	UHF Relay Radio (GFE)
Verify SS operating mode capabilities	X	X
Measure performance of the ranging channel:		
Signal-to-noise ratio	X	
Receiver sensitivity and lock-on range	X	
Error rate	X	
Modulation characteristics	X	
Measure performance of the command channel:		
Signal-to-noise ratio	X	
Receiver sensitivity and lock-on range	X	
Measure performance of the data channel:		
Signal-to-noise ratio	X	X
Modulation characteristics	X	
Signal distortion	X	X
Rf output power	X	
Rf output frequency and stability	X	
Receiver sensitivity and lock-on range		X
Accept composite command signal from Command SS OSE	X	
Provide TLM subcarrier to the TLM SS OSE	X	

RADIO SUBSYSTEM LEVEL TESTS Performance Parameter	S-Band Radio Equip-ment	UHF Relay Radio (GFE)	Inte-grated Tests
Receiver sensitivity (AGC Calib.)	X	X	
Ranging bandwidth	X		
Command bandwidth and output	X		
Data bandwidth and output	X	X	
Static phase error (SPE) vs. frequency	X		
Ranging and TLM modulation sensitivity	X		
Phase stability	X		
Frequency lock-on range	X	X	
Sweep-rate with lock-hold	X	X	
Ranging phase delay and loop Calib.	X		
Threshold loop bandwidth ($2B_{Lo}$)	X		
AGC loop bandwidth	X		
Receiver tuning range	X		
Incidental-phase-modulation (IPM)	X	X	X
Line-conducted-noise susceptibility	X	X	X
Rf power output	X		
EMI susceptibility & spurious outputs	X	X	X
VSWR	X		
Output freq. accuracy and stability	X		
VCO frequency	X		
Auxiliary oscillator frequency	X		
Telemetry monitor outputs	X	X	X
Control functions and redund. switching	X	X	X
Input power and voltage	X	X	X

1) Spacecraft Transmitter Exciter--Measure the frequency and power parameters and modulation characteristics of the spacecraft exciter; calibrate the exciter failure-sensing elements;
2) Spacecraft Transmitter TWTA--Measure frequency and power parameters of the TWTA; calibrate the TWTA failure-sensing elements;
3) Spacecraft S-Band and UHF Receivers--Measure the frequency and power parameters and demodulation characteristics; calibrate the failure-sensing elements;
4) Redundancy Control Unit--Calibrate the failure-sensing element monitor inputs; evaluate the control unit switching signal outputs;
5) Planetary Ranging Unit--Evaluate the performance of the ranging clock and code channel; simulate the in-lock signal input.

Signal Display--The SS OSE displays pertinent signal parameters and status, including time, alarm signals, signal waveforms, stimulus, and test data, spacecraft radio and SSTS status, power, current, voltage, frequency, spacecraft radio and SSTS operating mode, ranging error data, and data header information.

Test and Trend Data--The SS OSE records all test and trend data obtained during the subsystem TAT and FAT. Trend data consists of measurements of subsystem parameters that can be analyzed to provide a prediction of subsystem performance over a specified interval of operating time in a known or predicted environment. These parameters include: output rf power level, output frequency, and frequency stability, input power level, internal dc power supply voltage, receiver threshold level, and receiver lock-on range. In addition, fault-isolation test data are recorded by hand, as are measurements of rf spurious outputs, VSWR, rf signal loss calibration, phase stability, receiver threshold loop BW, AGC loop BW, incidental-phase-modulation, ranging-phase delay, frequency sweep rate with lock-hold, ranging and TLM modulation sensitivity, line-conducted-noise susceptibility, receiver tuning range, and rf power output calibration. Rf power output verification will be automatically recorded.

SSTS Self-Check--The SS OSE, in providing self-check capability, will measure and display stimulus outputs, simulate alarm conditions, measure power supply output voltage and current limiting, and simulate the spacecraft transponder with the test frequency converter and the spacecraft ranging simulator.

Functional Operation--The radio subsystem OSE simulates and displays the spacecraft radio input signal functions and parameters, and monitors, displays, records, and evaluates (with operator assistance) the derived output. Functional flow is shown in Figure 4.2-8.

The signal simulator provides: (1) control signals to switch the radio subsystem to all operating modes; and (2) command and telemetry signals, delivered to the S-band transponder and returned through the data evaluator, the programmer evaluator, or both.

Ranging signals are provided by the ranging test unit, delivered to the S-band transponder, and routed through the test receiver back to the ranging test unit, where they are correlated and evaluated.

The simulated and derived signals, along with frequency, voltage, and power measurements digitized by the measuring equipment, are transferred from the distribution unit to the data formatter, where the data are formatted for recording, supplemented with header identification, and then transferred to either the high-speed printer or the magnetic-tape recorder.

The time and clock generator provides time for operator display and data annotation, and clock signals for control of the data flow.

When operated with the STC, the SSTS is controlled by the programmer-evaluator, through a paper-tape reader controlled in turn by the STC CDCS. The programmer connects the proper simulator and measuring equipment, and the magnetic-tape recorder and data formatter. The programmer stops at the end of a particular test sequence or when an out-of-tolerance measurement indicates a fault. Fault isolation is accomplished manually. The programmer may also be used locally to control a number of the subsystem tests, but the primary test mode at the subsystem test level is manual.

The spacecraft ranging simulator and the test frequency converter together provide simulation of the spacecraft transponder for ranging phase-delay calibration and self-check of the S-band test transmitter and receiver.

Automatic power turn-off is effected by the alarm monitor panel, the VSWR monitor, or the power control when failure or incorrect switching causes excessive spacecraft or test set temperatures, currents, or rf VSWR's.

The relay radio link test equipment simulates the characteristics of the capsule signal and modulation, and measures the characteristics of the relay link receiving equipment.

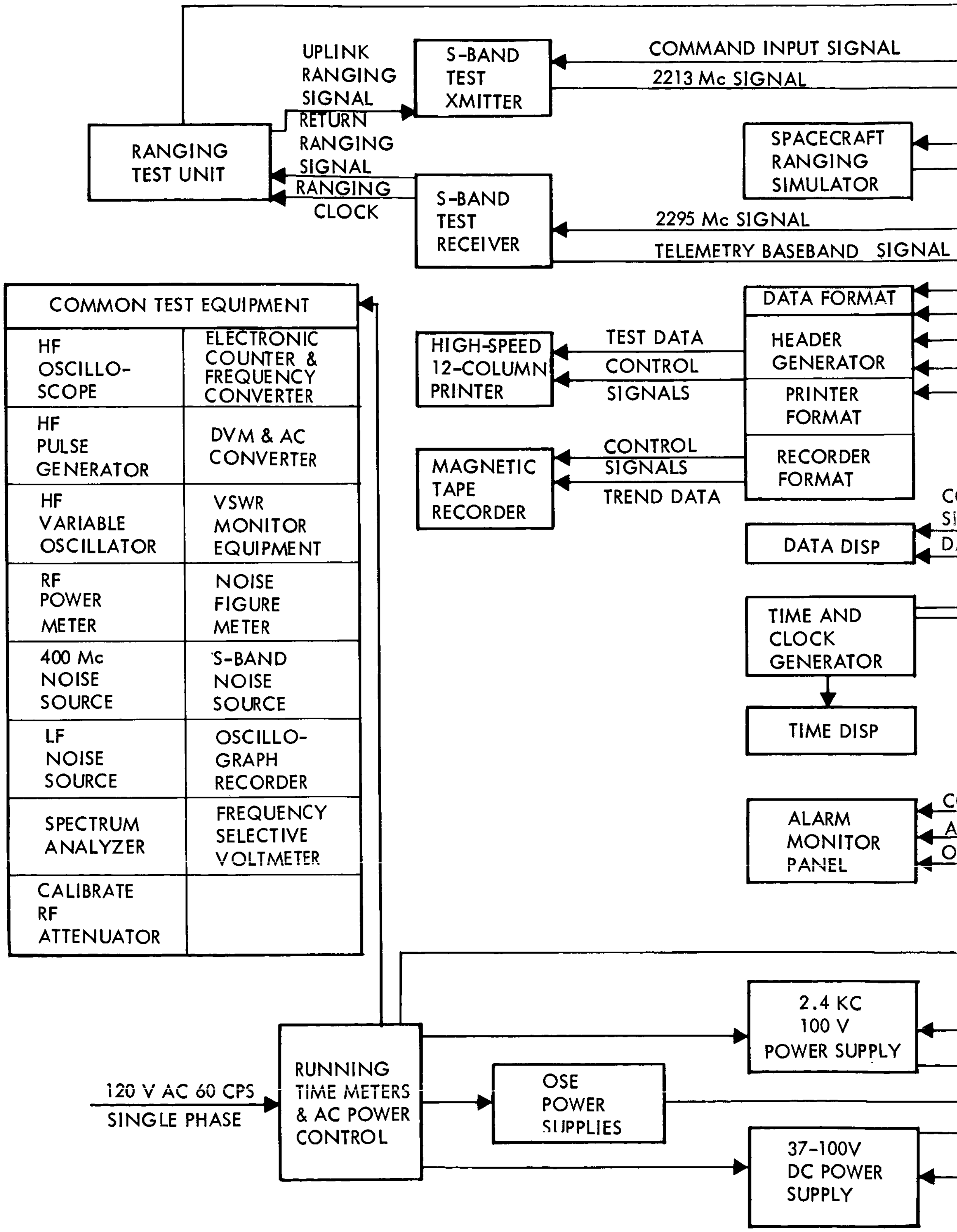
4.2.4.2 Interfaces

The radio subsystem OSE has the following interfaces

1) SS OSE with spacecraft radio subsystem
   a) The input and output signals (see Figure 4.2-8);
   b) The isolated OSE connector signals, which are C&S subsystem control signal inputs, telemetry monitor sensor outputs, test alarm sensor outputs, command subcarrier output, telemetry subcarrier input, internal d.c. power voltages, range code, and redundancy controls.
2) SS OSE with telemetry subsystem OSE--At this interface, the S-band test receiver will provide to the telemetry SSTS a detected composite telemetry signal on a subcarrier.
3) SS OSE with command subsystem OSE--At this interface, the command subsystem OSE will provide a composite signal for modulation of the S-band test transmitter.
4) SS OSE with STC adapter--The interfaces with the STC adapter are:
   a) Control lines to start test sequences, change mode status, turn power on and off, and operate safety interlocks;
   b) Signal lines to report OSE and spacecraft radio subsystem status, and to provide to the STC raw stimulus data and raw and processed response data.

4.2.4.3 Physical Characteristics

The radio subsystem OSE will be housed in three enclosures of two integrally connected standard NASA/JPL racks each, one enclosure being reserved for the relay radio link test equipment. In addition, two integrally connected test consoles will contain the display and control equipment. Total SS OSE weight will be approximately 5200 pounds to



UPLINK RANGING SIGNAL
S-BAND TEST XMITTER
COMMAND INPUT SIGNAL
2213 Mc SIGNAL
RETURN RANGING SIGNAL
RANGING TEST UNIT
RANGING CLOCK
SPACECRAFT RANGING SIMULATOR
S-BAND TEST RECEIVER
2295 Mc SIGNAL
TELEMETRY BASEBAND SIGNAL
COMMON TEST EQUIPMENT
HF OSCILLO-SCOPE
ELECTRONIC COUNTER & FREQUENCY CONVERTER
HF PULSE GENERATOR
DVM & AC CONVERTER
HF VARIABLE OSCILLATOR
VSWR MONITOR EQUIPMENT
RF POWER METER
NOISE FIGURE METER
400 Mc NOISE SOURCE
S-BAND NOISE SOURCE
LF NOISE SOURCE
OSCILLO-GRAPH RECORDER
SPECTRUM ANALYZER
FREQUENCY SELECTIVE VOLTMETER
CALIBRATE RF ATTENUATOR
HIGH-SPEED 12-COLUMN PRINTER
TEST DATA
CONTROL SIGNALS
DATA FORMAT
HEADER GENERATOR
PRINTER FORMAT
RECORDER FORMAT
MAGNETIC TAPE RECORDER
CONTROL SIGNALS
TREND DATA
DATA DISP
TIME AND CLOCK GENERATOR
TIME DISP
ALARM MONITOR PANEL
2.4 KC 100 V POWER SUPPLY
120 V AC 60 CPS SINGLE PHASE
RUNNING TIME METERS & AC POWER CONTROL
OSE POWER SUPPLIES
37-100V DC POWER SUPPLY

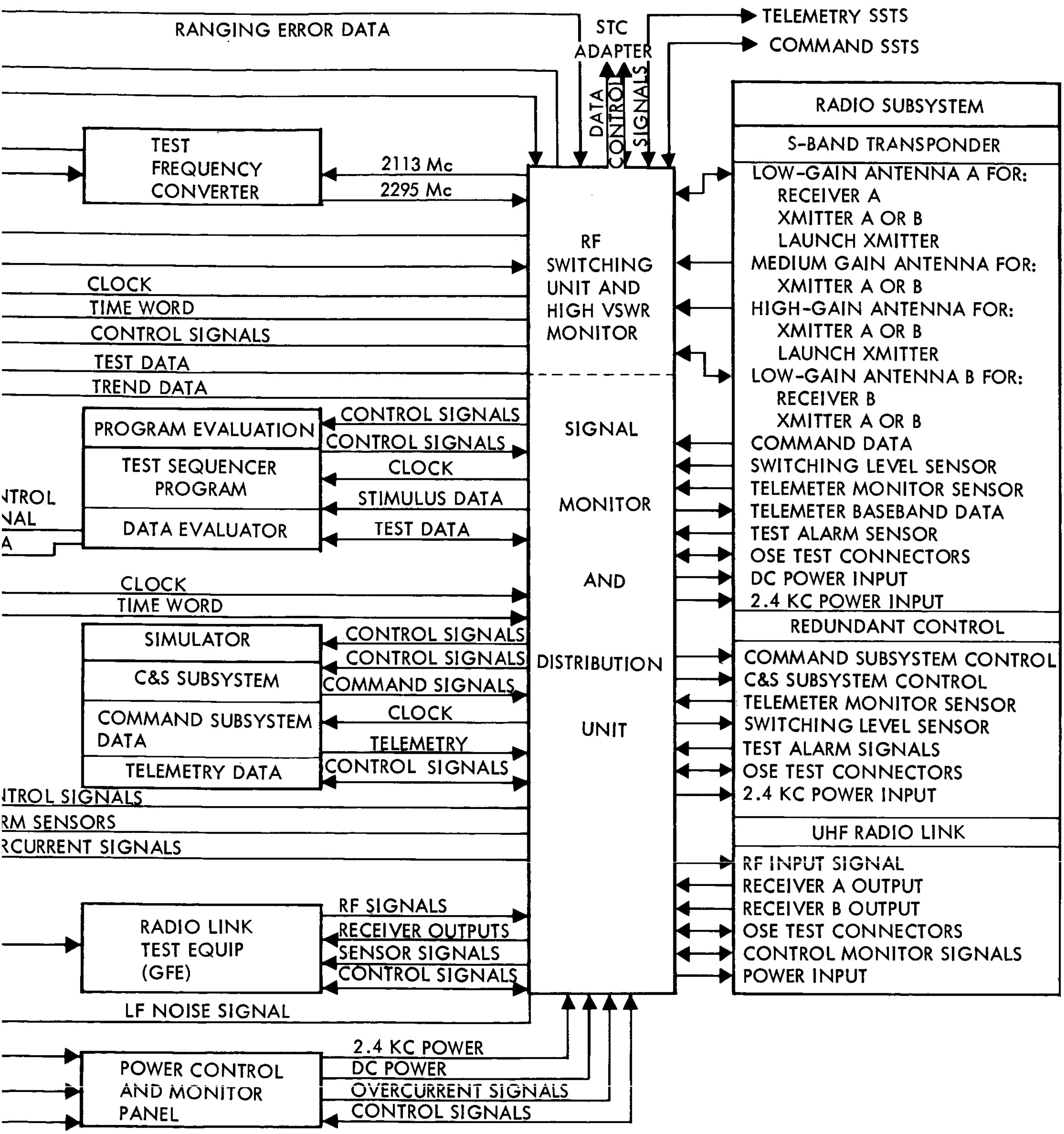


Figure 4.2-8: Radio Subsystem OSE Functional Flow

2

1500 pounds for each enclosure, and 700 pounds for the double test console enclosure.

The power requirement for the OSE will be 104-125 volts, 55-65 cps, single phase at approximately 6200 watts.

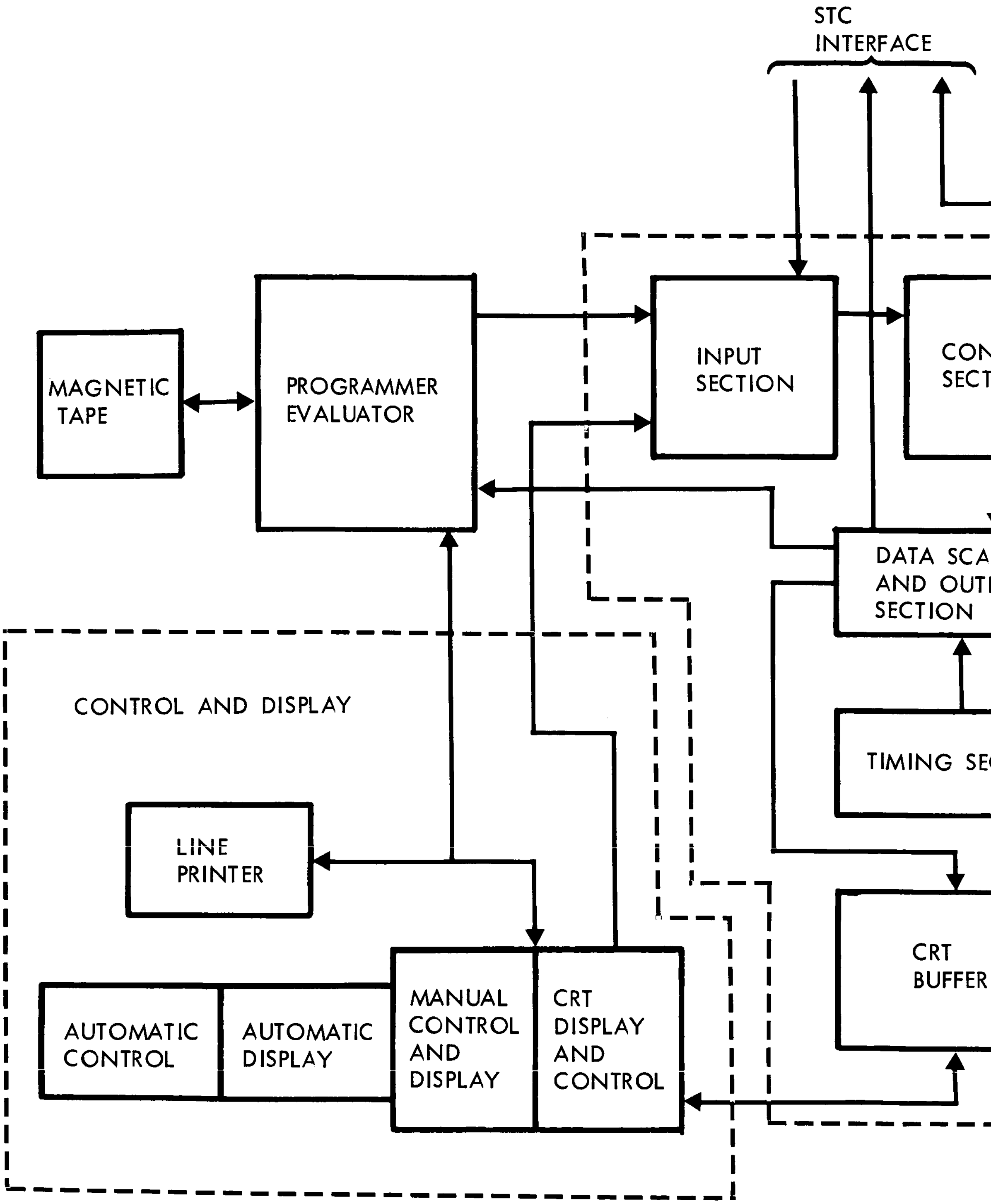
4.2.5 Telemetry Subsystem OSE

The telemetry subsystem OSE is used for bench testing during initial performance verification of the telemetry subsystem and also during subsystem integration into the Flight Spacecraft. During bench testing, the OSE is controlled by the OSE programmer and during system integration tests by the STC CDCS. The telemetry subsystem OSE provides power and sync signals, mode control commands, and simulated data signals to the telemetry subsystem. The telemetry subsystem OSE performs subcarrier demodulation, and data decommutation, and provides visual display and readout of the data for verification of proper telemetry subsystem performance.

4.2.5.1 Functional Description

The block diagram of the preferred telemetry subsystem SS OSE design is shown in Figure 4.2-9. The programmer-evaluator provides overall control by means of a stored program. These internal control signals control the actual generation of the data by the input-output buffer and the auxiliary test equipment. Control is also exercised over the data returned from the telemetry subsystem by the system test telemetry data decommutator. The interpreted results are routed through a CRT buffer and displayed along with monitor and status data. The data are also routed to a line printer for a hard-copy record, and selected trend data are

B L A N K


STC
INTERFACE
MAGNETIC
TAPE
PROGRAMMER
EVALUATOR
INPUT
SECTION
CON
SECT
DATA SCA
AND OUT
SECTION
TIMING SE
CONTROL AND DISPLAY
LINE
PRINTER
AUTOMATIC
CONTROL
AUTOMATIC
DISPLAY
MANUAL
CONTROL
AND
DISPLAY
CRT
DISPLAY
AND
CONTROL
CRT
BUFFER

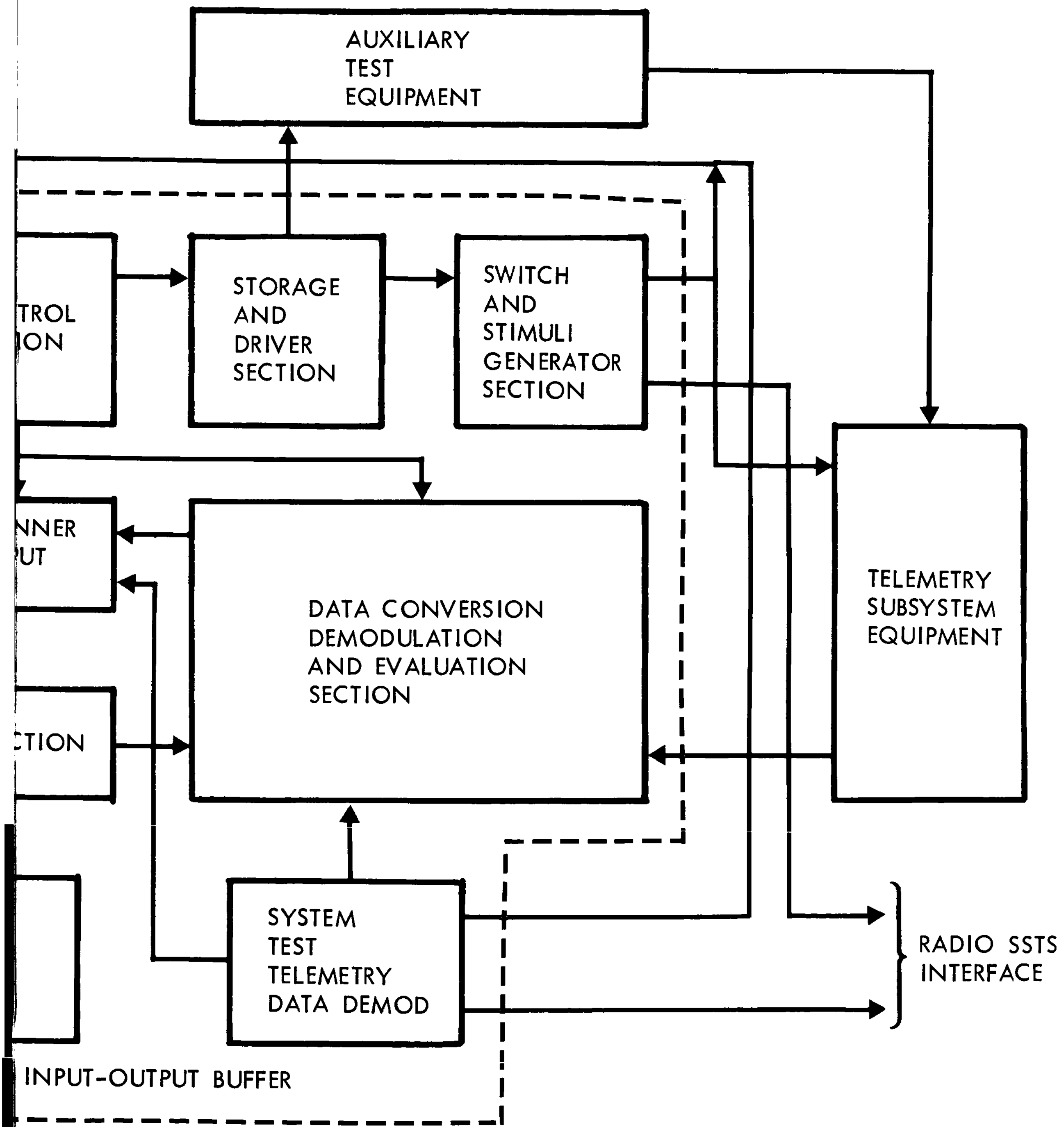


Figure 4.2-9: Telemetry SSTS Functional Block Diagram

recorded on a magnetic tape. The main difference between this approach and that used on Mariner is the greater emphasis on automatic programming and evaluation for Voyager.

Automatic programming and evaluation is used because of: (1) a greater task accomplishment ability, (2) greater functional reliability, (3) greater cost-effectiveness, and (4) flexibility. It is estimated that a single run at normal voltages and temperatures may include as many as 2500 measurements, and require up to 40 hours if performed manually, but only 4 hours when performed automatically at a rate of one measurement per second. In light of these factors, the use of fully automatic techniques using a computer was selected in spite of the higher initial costs.

Automatic, semiautomatic, or manual modes are provided for bench subsystem testing. In the automatic mode, all the tests proceed under the direction of the computer. Status and test results are printed out by a line printer, displayed on the CRT, and recorded on magnetic tape.

In the semiautomatic mode, operator selected tests are looped, but run under computer control within the loop. The status and test results are recorded and displayed as in the automatic mode except that the magnetic tape will not be prepared.

In the manual mode, data are still cycled through the telemetry master digital multiplexer and block encoder, and selected words are displayed on an analog CRT for operator use. The computer is not active in this mode except as necessary to generate the data. In this mode, the application of stimuli and the measurement and interpretation of the results are at the discretion of the operator.

The input-output buffer (IOB) takes a control word from either the local programmer-evaluator or the STC CDCS and converts this control into the proper stimulus to the telemetry subsystem. This may cover the range from a single bit designating a mode command to many thousands of bits in a stream of data. The inputs provided by the IOB are designed to simulate the normal inputs to the telemetry subsystem as closely as possible. Similarly, the IOB also simulates the output loading on the telemetry subsystem and processes the data into a form suitable for display or delivery to the local or STC computer.

The diversity of signals and the functions that the IOB handles dictates the use of special-purpose equipment. The CRT buffer is special-purpose equipment.

The various data display and control requirements, and the various media used to meet these requirements are summarized in Table 4.2-5.

4.2.5.2 Interfaces

The interfaces for the SSTS consist of the following

1) Telemetry subsystem interfaces--See Section 4.1.5 of Volume A.
2) Data and control with the STC.
3) Demodulated serial PCM telemetry bit stream from the radio SSTS.
4) "Real-time data" and bit sync to the science SSTS.

4.2.5.3 Performance Parameters

The telemetry subsystem OSE provides the following functions during subsystem bench tests and during system integration tests:

TABLE 4.2-5: SUMMARY OF DISPLAY CONTROL REQUIREMENTS
(X DESIGNATES A REQUIREMENT)

DISPLAY MEDIA	BENCH SUBSYSTEM TESTS			INTEGRATION STC TESTS	TYPE OF DATA
	Auto. Test Mode	Semi-Auto. Mode	Manual Diagnosis	Auto. Test Mode	
In-line Alphanumeric Projection	X	X	X		Time, analog channel voltages
Bilevel Indicators	X	X	X	X	Digital channel selection, Test Set & Computer status
Alphanumeric Cathode ray tube (CRT)	X	X	X		Input analog, input data, & digital data channel parallel display
Analog CRT		X	X		Telemetry analog output, radio SSTS input
Switch entry	X	X	X	X	Mode Control. Subroutine selection. Time entry.
Line printer	X	X			Hard copy of all other displays
Alarms, visual and aural	X	X	X	X	Voltage, temperature, discrete errors.

Bench Tests

1) Power and Timing Inputs--The telemetry subsystem OSE supplies power in the form of 100-volt peak-to-peak, 2.4-kbps squarewave to the lower and upper subcarrier power supplies in a redundant bus configuration. In addition, a 460.8-kc clock train is supplied for the lower and upper subcarrier lock oscillators.

2) Subsystem Command and Control--The telemetry subsystem OSE provides signals to simulate the mode control normally supplied by the spacecraft computer and sequencer subsystem and the command subsystem. These signals are described in Section 4.1.5 of Volume A on the telemetry subsystem.

3) Data and Status--Provide signals to simulate the inputs normally supplied by the data storage subsystem, as follows:

    a) Digital data in serial binary-coded format at 11.5, 15.4, 55.4, 1200, and 7200 bps. This covers the simulation of planetary-science data, capsule data, maneuver data, and flare data.

    b) Record and reproduce status.

    c) Clock signals for data synchronization.

    d) Record engineering data including pressure and temperature.

    e) Supply digital data to the master digital multiplexer to simulate the signals normally supplied by the Science Subsystem, the capsule relay, the command subsystem, and the computer and sequencer. Supply data for the engineering data multiplexer/encoder for up to 300 analog channels and 150 digital channels.

4) Monitor and Display the following telemetry subsystem functions.

    a) Check the response of the redundant power supplies to input power transients including loss of one input bus.

b) Verify the operation of the countdown logic, the bandpass filters, and the bandpass filter switch by measuring the subcarrier output frequencies for the lower and upper subcarrier channels.

c) Verify the correct operation of the frame sync logic, the word ID logic, and the frame count logic by exercising the mode controls and monitoring the output of the master digital multiplexer.

d) Verify the activation of the mode commands and proper identification bits in the output words.

e) Verify correct operation of the biphase modulators and the summing networks by measuring the output levels, the power spectral density, and the phase shift characteristics.

f) Verify the correct operation of the master digital multiplexer by measuring the output in response to simulated inputs on the cruise science data channel, the flare and maneuver data channel, and the capsule engineering data channel.

g) Define the input-output characteristics of the engineering multiplexer telemetry channels. For the analog channels, the following parameters are measured: (1) conversion accuracy, (2) cross-talk, (3) linearity, (4) conversion time, and (5) gross channel independence. For the digital channels, the following parameters are measured: (1) absolute bit content, (2) noise immunity, and (3) gross channel independence.

h) Verify the proper operation of the various buffer storage logic units by correlating the output with simulated inputs on the following channels: (1) C&S subsystem, (2) command subsystem, and (3) flare and maneuver data channel.

i) Verify proper operation of the block encoder.

j) Additional fault isolation tests will be incorporated as the telemetry subsystem detail design progresses.

5) The test set edits the test data and records on magnetic tape data selected for later trend analysis.

During Integration Tests

1) The OSE is capable of monitoring the inputs and outputs of the telemetry subsystem corresponding to Items 1 through 4 under "Bench Tests," above. This includes the voltage level, waveform characteristics, and any degradation due to noise.

2) The monitoring required in the integrated system test affords fewer test points compared to the bench tests because the normal spacecraft cabling cannot be disturbed. The outputs must be monitored by high-impedance devices ($Z_{in} \leq 1$ percent of R_L) to prevent upsetting the normal interface relations of the spacecraft subsystems.

3) Direct access to raw test data is provided for the STC.

4) A means of controlling the application, sequence, and selection of the system-level test stimuli by the STC is provided.

5) A means will be provided to indicate telemetry subsystem OSE status to the STC.

6) Receives the demodulated rf carrier from the radio SSTS.

4.2.5.4 Physical Characteristics

The telemetry subsystem OSE is contained in approximately five standard NASA/JPL racks. The functional elements are shown in Figure 4.2-9.

4.2.5.5 Trade Studies

A summary of the telemetry trade studies is shown in Table 4.2-6.

4.2.6 Command Subsystem OSE

The command subsystem OSE supports checkout and fault isolation of the command subsystem to the component level. It supports the TAT and the FAT of the command subsystem and operates in conjunction with other subsystem OSE and the STC to perform the spacecraft system testing.

The Voyager command testing differs from the Mariner C in that the testing is accomplished with a separate set of OSE as opposed to the test capability being included as a part of the radio subsystem for Mariner. The Voyager command OSE is more extensive since full decoding capability is included in the Voyager command subsystem.

4.2.6.1 Functional Description

The spacecraft command subsystem OSE simulates and displays the command subsystem input signal functions and parameters. It monitors, displays, records and, with operator assistance, evaluates the derived outputs. A functional flow diagram of the command subsystem OSE is shown in Figure 4.2-10.

Major Elements--The command subsystem OSE includes the following major elements.

1) Power--Power the command subsystem during test with 50-volt rms ± 15 percent, 2.4-kc squarewave power.

TRADE STUDY SUMMARY SHEET — SOURCE OF REQUIREMENT Section 4.0 (SS OSE) — FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	TRADE STUDY NUMBER & TITLE: TELEMETRY SUBSYSTEM OSE TRADE STUDY SUMMARY — MATRIX OF DESIGN APPROACH — MANUAL CONFIGURATION (Telemetry Subsystem OSE)	SEMIAUTOMATIC CONFIGURATION (Telemetry Subsystem OSE)	AUTOMATIC CONFIGURATION (Telemetry Subsystem OSE)	SELECTION: ✓ Manual	✓ Semi-Automatic	✓ Automatic
Functions that must be performed by the Telemetry Subsystem OSE.	All checkout operations performed by the OSE operator.	OSE operator must assist the Subsystem OSE to perform each subsystem test.	No assistance required Subsystem OSE performs complete subsystem test routing.			
1. A programmer must serve as a source for command and control signals that are used to drive the S/C telemetry and OSE simulation equipments.	1) Could be done manually	1) Could be done semiautomatic	1) Can be made automatic	✓	✓	✓
2. Evaluation of the command and control signals supplied by the telemetry subsystem to the data storage subsystem tape recorder.	2) Could be done manually	2) Could be done semiautomatic	2) Can be made automatic	✓	✓	✓
3. Simulation of data signals normally supplied by the data storage subsystem.	3) Could <u>not</u> be done manually	3) Could be done semiautomatic	3) Can be made automatic		✓	✓
4. Simulation of data signals normally supplied by the science subsystem and the capsule engineering data equipment.	4) Could <u>not</u> be done manually	4) Could be done semiautomatic	4) Can be made automatic		✓	✓
5. Evaluation of PCM output data of telemetry subsystem.	5) Could <u>not</u> be done manually	5) Could be done semiautomatic	5) Can be made automatic		✓	✓
6. Simulation of spacecraft engineering data.	6) Could <u>not</u> be done manually	6) Could be done semiautomatic	6) Can be made automatic		✓	✓
7. Evaluating output of the multiplexer/encoder in response to test data.	7) Could <u>not</u> be done manually	7) Could be done semiautomatic	7) Can be made automatic		✓	✓
8. Storing a program for overall evaluation and control.	8) Could <u>not</u> be done manually	8) Could be done semiautomatic	8) Can be made automatic		✓	✓
9. Processing of control data generated by the manual functions under control of the operator and processing of display data.	9) Could be done manually	9) Could be done semiautomatic	9) Can be made automatic	✓	✓	✓
10. Selecting, sorting, and formatting status and trend data for magnetic recording.	10) Could <u>not</u> be done manually	10) Could be done semiautomatic	10) Can be made automatic		✓	✓
11. Format data for line printer, hard copy.	11) Could <u>not</u> be done manually	11) Could be done semiautomatic	11) Can be made automatic		✓	✓
12. Self-check closed loop operation.	12) Could <u>not</u> be done manually	12) Could be done semiautomatic	12) Can be made automatic		✓	✓
				3	12	12 ◁•▷
	Manual operation must consider high error rate, due to complexity of test, on part of manual operator	Semiautomatic operation reduces operator error rate, but does not eliminate errors.	Automatic operation reduces all operator error.	SELECTED APPROACH: Trade study results indicate that the Telemetry Subsystem OSE configuration must include an automatic mode of operation.		

◁•▷ A JPL OSE requirement states that all extensive or complex subsystem test routines must be performed expeditiously, correctly, and repetitively. This statement requires that a greater weight be placed on each check mark in the automatic column.

TABLE 4.2-6: Telemetry Subsystem OSE Trade Study Summary

2) Control--Control the test operations by selecting input stimuli, measuring equipment, displays, and recording equipment. When the SS OSE is operated in the STC, the control of the SS OSE is from the tape reader with STC supplying test selection.
3) Stimuli--Generate composite command signals as directed by the programmer.
4) Displays--Display signal parameters and status to include time, alarm signals, signal waveforms, stimulus and test data, command subsystem and SS OSE status, power, current, voltage, frequency, command subsystem and SSTS operating mode, command error data, and data header information.
5) Recording--Record test data to include bit error rate at a specific signal-to-noise ratio, input power level, internal dc power supply voltages, decoder output signal jitter, detector signal level for lock, and detector threshold level.
6) Self-Check--Self-check the SS OSE as follows: measure and display the stimulus signals, simulate alarm conditions, and measure power supply output voltages and current limiting.

Test Sequence--Composite command signals of the form CPN $\oplus$ $2f_s$ are supplied by the signal simulator to the command subsystem, where they are detected and decoded, and to the data evaluator portion of the programmer evaluator. The command decoder has a control line output for each Voyager command. All control outputs are brought to the data evaluator for verification of proper control-line response versus generated command word.

The signal simulator also generates a 4.8-kc train of pulses to the command subsystem in response to a "data ready" signal from the command subsystem. The pulse train shifts out the partially decoded DAE or C&S subsystem command, depending on which "data ready" signal was received. The partially decoded command is transferred to the data evaluator for comparison with the generated command. Either the derived or generated command may be displayed.

The composite command signal may be summed with noise to permit evaluation of the command subsystem when receiving a signal with a low signal-to-noise ratio.

The simulated and derived command signals, as well as frequency, voltage, and power measurements digitized by the measuring equipment, are transferred to the data formatter. At the formatter, the data are formatted for recording, data identification header added, and the data transferred to either the high-speed printer or the magnetic-tape recorder.

Time for operator display and data annotation and clock signals for control of the data flow originate in the time and clock generator.

Control of the test set, when operated with the STC in the system test mode, is provided by the programmer in the programmer-evaluator. Control is provided by paper-tape reader so the system tests can be remotely and centrally controlled by the STC computer. The computer initiates each test sequence by advancing the tape reader. By virtue of the tape program, the programmer connects the proper simulator and measuring equipment, and the magnetic-tape recorder and data formatter. The programmer stops automatically at the end of each test sequence. Fault isolation is accomplished manually. The programmer may also be used locally to control a number of the subsystem tests, but the primary subsystem test mode is manual.

Operator warning and automatic power turn-off by the alarm monitor panel and the power control and monitor panel occur when failures or incorrect switching causes excessive temperatures or currents in either the command subsystem or the SS OSE.

4.2.6.2 Interface Definitions

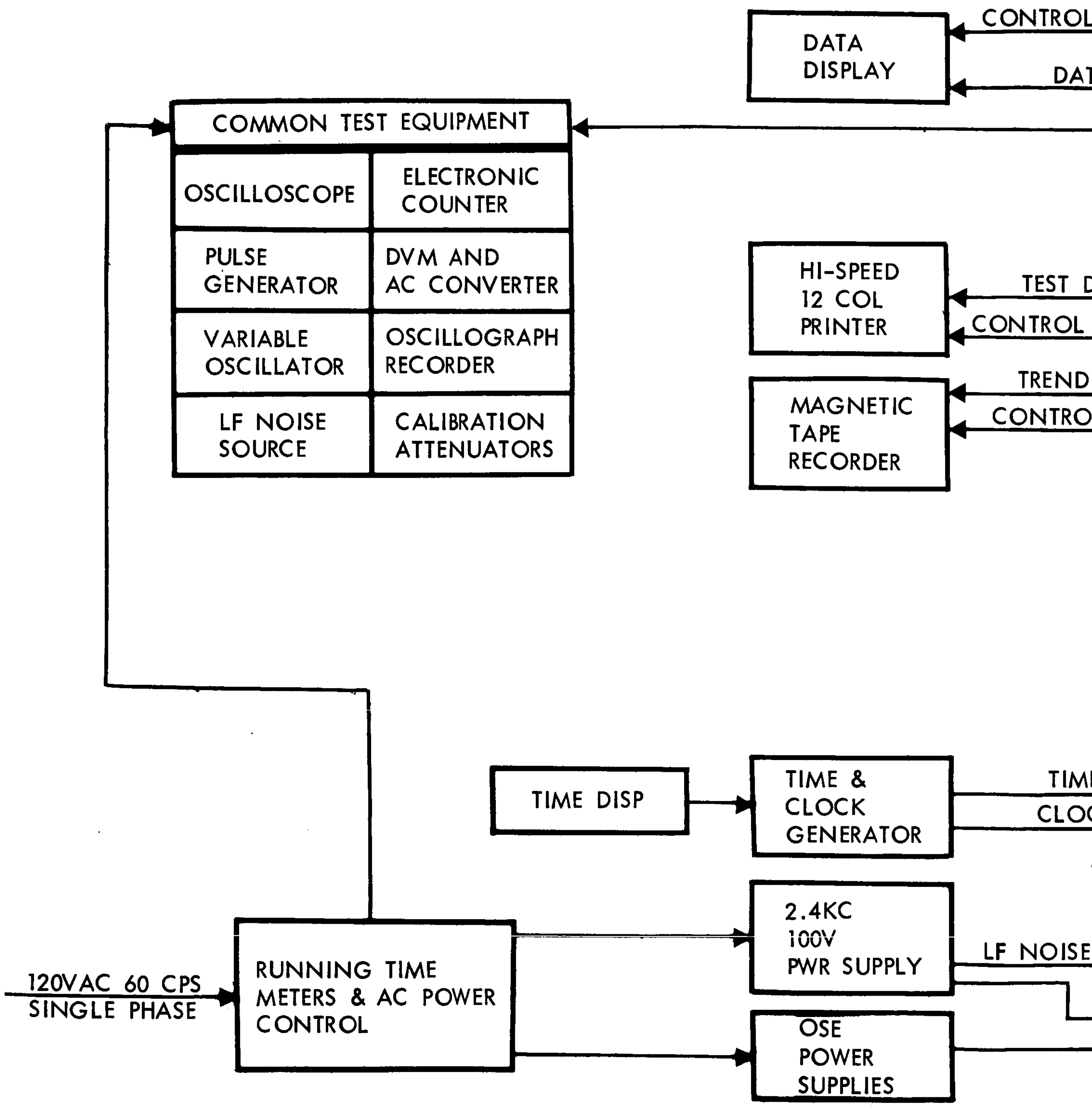
The external interfaces to the command subsystem OSE are as listed below. Spacecraft Command Subsystem--The interfaces with the command subsystem are: the input-output signals (see Figure 4.2-10) and the isolated OSE connector signals, which are:

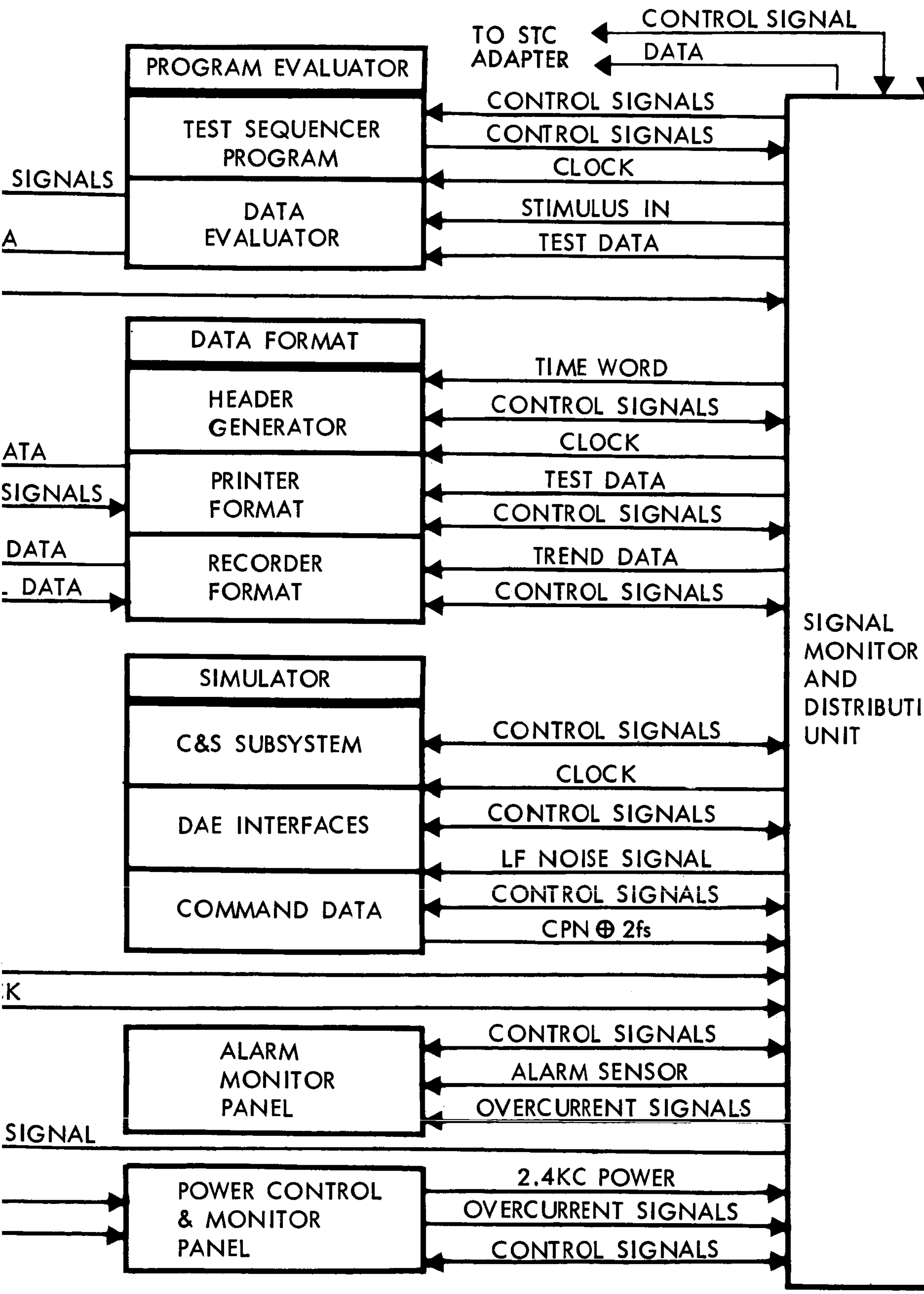
1) Detector VCO output;
2) Decoder internal clock output;
3) Telemetry monitor sensor outputs;
4) Decoder execute signal;
5) Decoder command verified signal;
6) Decoder parity correct signal;
7) Test alarm sensor outputs;
8) Command composite subcarrier input;
9) Internal dc power voltages.

Radio Subsystem OSE--The interfaces with the radio subsystem OSE are: composite command signal to the radio subsystem OSE for modulation on the S-band test transmitter, and composite command signal from the radio subsystem OSE for transfer to the command subsystem.

STC Adapter--The interfaces with the STC adapter are: control lines to start test sequences, change modes, turn power on and off, and to operate safety interlocks; signal lines to report SS OSE and command subsystem

B L A N K


DATA DISPLAY
CONTROL
DAT
COMMON TEST EQUIPMENT
OSCILLOSCOPE
ELECTRONIC COUNTER
PULSE GENERATOR
DVM AND AC CONVERTER
VARIABLE OSCILLATOR
OSCILLOGRAPH RECORDER
LF NOISE SOURCE
CALIBRATION ATTENUATORS
HI-SPEED 12 COL PRINTER
TEST D
CONTROL
MAGNETIC TAPE RECORDER
TREND
CONTRO
TIME DISP
TIME & CLOCK GENERATOR
TIM
CLOC
2.4KC 100V PWR SUPPLY
LF NOISE
120VAC 60 CPS SINGLE PHASE
RUNNING TIME METERS & AC POWER CONTROL
OSE POWER SUPPLIES


TO STC ADAPTER
CONTROL SIGNAL
DATA
PROGRAM EVALUATOR
TEST SEQUENCER PROGRAM
DATA EVALUATOR
SIGNALS
CONTROL SIGNALS
CONTROL SIGNALS
CLOCK
STIMULUS IN
TEST DATA
DATA FORMAT
HEADER GENERATOR
PRINTER FORMAT
RECORDER FORMAT
ATA
SIGNALS
DATA
DATA
TIME WORD
CONTROL SIGNALS
CLOCK
TEST DATA
CONTROL SIGNALS
TREND DATA
CONTROL SIGNALS
SIMULATOR
C&S SUBSYSTEM
DAE INTERFACES
COMMAND DATA
CONTROL SIGNALS
CLOCK
CONTROL SIGNALS
LF NOISE SIGNAL
CONTROL SIGNALS
CPN ⊕ 2fs
ALARM MONITOR PANEL
CONTROL SIGNALS
ALARM SENSOR
OVERCURRENT SIGNALS
SIGNAL
POWER CONTROL & MONITOR PANEL
2.4KC POWER
OVERCURRENT SIGNALS
CONTROL SIGNALS
SIGNAL MONITOR AND DISTRIBUTI
UNIT

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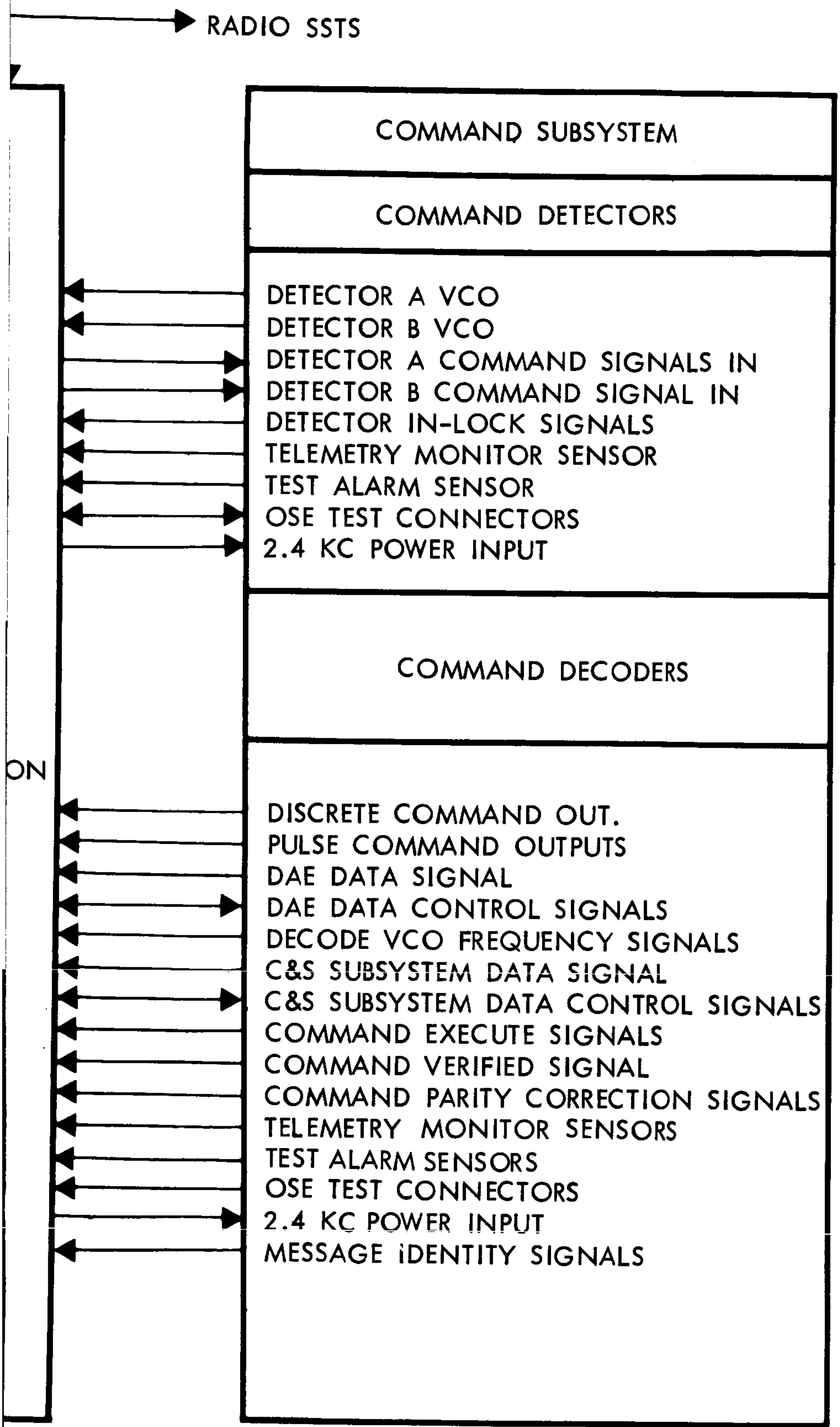


gure 4.2-10: Command Subsystem OSE Functional Flow

status; signal lines to provide raw stimulus data to the STC; and signal lines to provide raw and processed data to the STC.

4.2.6.3 Performance Parameters

The SS OSE evaluates the command subsystem performance at the various levels of test as delineated below.

1) System Level
   a) Measure the performance of Command Channels A and B.
   b) Verify the ability of Command Channels A and B to supply commands to the C&S subsystem and the DAE.
   c) Verify the ability to decode and execute all command functions.

All subsystem functions can be verified either in the presence of controlled levels of noise in the command signal or without noise.

2) Subsystem Level
   a) Verify the calibration of the telemetry monitor sensors.
   b) Measure the frequency and power parameters of the subsystem.
   c) Determine the susceptibility to line-conducted noise.
   d) Verify the correct data pulse outputs to the telemetry subsystem.
   e) Verify the correct failure-logic switching within the subsystem.
3) Fault Isolation
   a) Passive Analog Mixer
      (1) Measure the frequency and power parameters of the detector.
      (2) Determine the ability to detect and synchronize, in the presence of noise, a composite command signal of the form CPN $\oplus$ $2f_s$.
      (3) Verify the ability to provide the VCO frequency and in-lock signal to the telemetry subsystem.

b) Command Detector

(1) Measure the frequency and power parameters of the detector.

(2) Determine the ability to detect and synchronize, in the presence of noise, a composite command signal of the form CPN $\oplus$ $2f_s$.

(3) Verify the ability to provide the VCO frequency and in-lock signal to the telemetry subsystem.

c) Detector Selection Logic

(1) Measure the power parameters of the logic unit.

(2) Determine the ability to pass, without distortion, the detected command signal.

d) Command Decoder

(1) Measure the frequency and power parameters of the decoder.

(2) Verify the capability to decode and synchronize detected command signals.

(3) Verify the ability to provide the parity correct, command verify, message destination, and execute signals to the telemetry subsystem.

4.2.6.4 Physical Description

The command subsystem OSE will be housed in one enclosure of two integrally connected standard NASA/JPL racks, and one standard NASA/JPL rack. In addition, two integrally connected test consoles will contain the display and control equipment. The weight of the double rack enclosure will be approximately 1500 pounds, and approximately 850 pounds for the single rack. The double test console will weigh approximately 750 pounds for a total OSE weight of approximately 3100 pounds. The power requirement for the OSE will be 105-125-volts, 55-65-cps, single phase at approximately 2700-watts.

4.2.7 Computer and Sequencer Subsystem OSE

The computer and sequencer SS OSE tests the subsystem operation at two functional levels: diagnostic and operational. The diagnostic testing involves malfunction detection and fault isolation down to the replaceable subassembly. The operational testing simulates mission sequences under environmental stress to verify the subsystem. These tests are performed on the subsystem during manufacturing, TAT, and FAT testing. The SS OSE also executes portions of the STC and LCE controlled tests.

The design objective of the SS OSE is to provide automatic testing to eliminate human errors in repetitious testing of many signals and to provide self-check without disconnecting from the subsystem. Sufficient flexibility of test equipment is provided to ensure a fast response to failures during manual or semiautomatic tests. The design of the advanced computer and sequencer subsystem OSE developed by NASA/JPL and documented in NASA/JPL Space Program Summary 37-33, Volume II has been used in developing the preferred design.

4.2.7.1 Functional Descriptions

The computer and sequencer SS OSE functional blocks are shown in Figure 4.2-11 and defined in Figure 4.2-12.

The computer and sequencer SS OSE uses seven major functional elements:

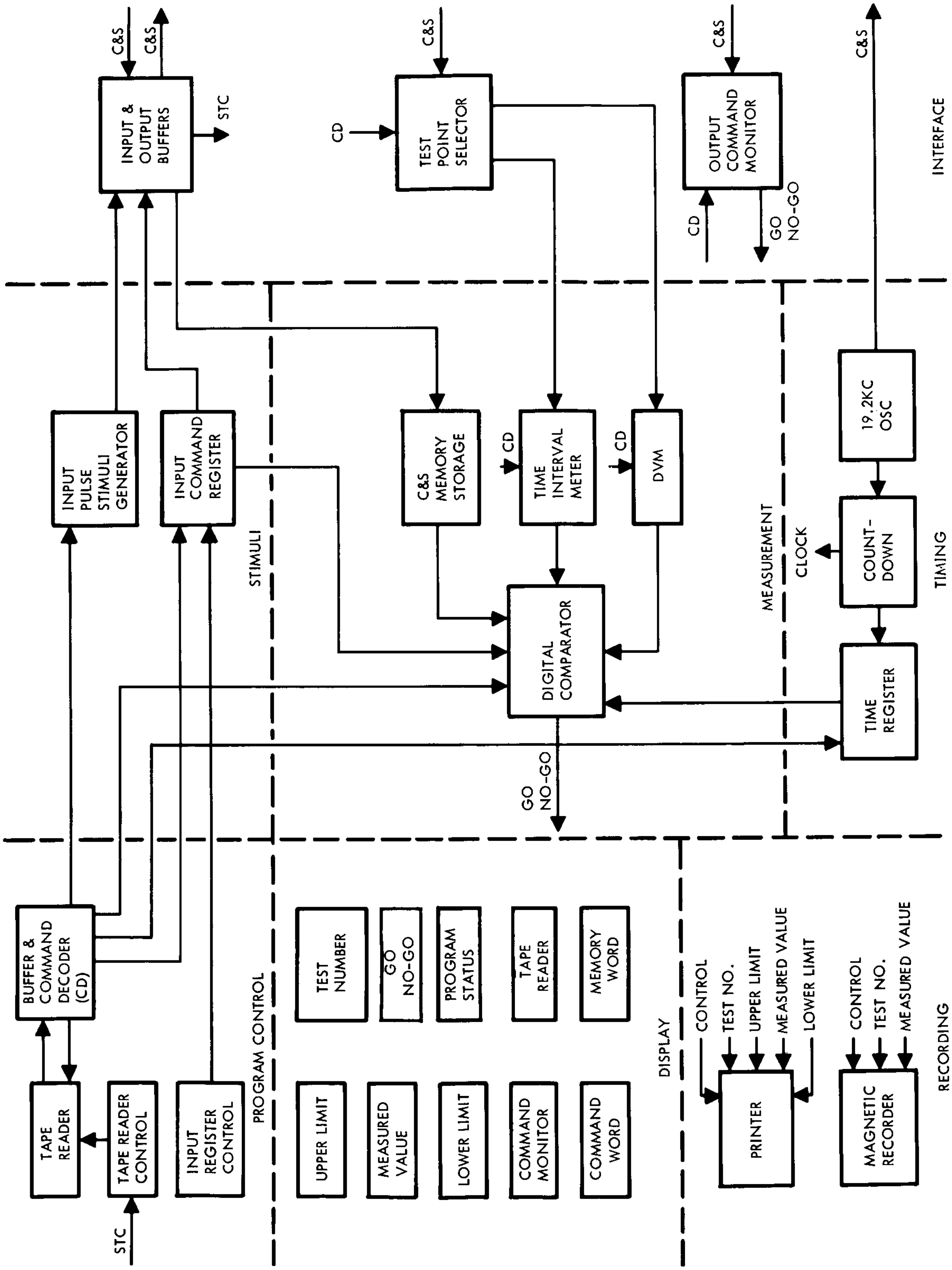


Figure 4.2-11: Computer And Sequencer Subsystem OSE

PROGRAM CONTROL SECTION		STIMULI SECTION		MEASUREMENT SECTION		RECORDING SECTION		DISPLAYS SECTION		TIMING SECTION		INTERFACE SECTION	
Tape Reader	Transport and reading of stored program from punched tape	Input Command Register	A 28-bit command word is loaded into the register from either the tape program or manual switches. The register is clocked out to the C&S.	C&S Memory Storage	Contents of any C&S memory location word can be transferred to this register for comparison to the programmed word stored in the input register from the tape reader.	Printer	Records test number, upper and lower limit measured value no-go flag, for local evaluation. Printer Modes include print all test data, or print test data on no-go only. Command decoder provides control.	Test Number	Decimal display of test being performed.	Reference Oscillator	Provides 19.2-kc reference signal to the C&S and to the test set countdown chain.	Buffers	Input-output buffers provide signal isolation and line drivers between the test set and the C&S for digital signals.
Tape Reader Control	Modes 1. Automatic: Proceed according to program. 2. STC control: Read and execute test number on direction. 3. Recycle: Read and execute one test at a time, repetitively. 4. Search Forward/Reverse: Manual insertion of test number with test search to provide random access to tests. 5. Single Frame Advance: Advance one frame at a time for trouble-shooting.	Pulse Generator	Provides discrete command pulses or pulse trains on command from the program. The simulated accelerometer pulses are controlled in rate and number of pulses. Transfer to the C&S is nonsynchronous.	Time Interval Meter	Evaluates pulse characteristics of rise time, fall time and pulse width and pulse spacing. Digital outputs are compared to limit values in the comparator.	Magnetic Recorder	Provides a record for remote evaluation and trend analysis. Records test number and measured value Command decoder provides control.	Measured Value	Decimal display of numerical value of measurement.	Countdown Chain	Frequency dividers from the oscillator provide the clock for operation of the test set and the input to the test set time accumulator.	Test Point Selection	Provides the selection of the output C&S commands to be measured by the test set. Selection is controlled by the command decoder from the program. Simulated loading is included.
Buffers	Buffers receive parallel output of tape reader for transfer to the command decoder or input register.			Digital Voltmeter	Measures amplitudes of pulse and discrete signals and power supply voltages. Digital comparison to limits in comparator.			Upper/Lower Limit	Decimal display of programmed limits for measurement being made.	Time Register	Provides the accumulative time to compare to the time register in the C&S. The test set time register as well as the C&S time register can be advanced to any state to reduce the wait periods in the mission sequence.	Output Command Monitor	Provides continuous monitoring of all command lines to detect the occurrence of any undesired commands. Presence of a command on any line (except programmed command) will set an indicator. Desired command indicator is inhibited by the command decoder. Presence of an undesired command will stop the program. Visual indicators on the display panel will show which command occurred.
Command Decoder	Command decoder translates tape reader output to execute command for the test set. Selection and control of measurement device, test point, stimuli generators, time register set, command monitor inhibit, comparators, printers, and recorders.			Digital Comparator	Makes a go/no-go decision based on comparison of the measured value to the programmed limits.			Go-No Go	Indicates acceptance or rejection of test value.				
Input Register Control	Individual bit set-reset for manual inserting a word for transfer to C&S memory.							Program Status	Indicator lamps showing program stop, momentary hold, test recycle, parity error (tape reader) parity error (word transfer)				
								Tape Reader	Binary display of tape reader output.				
								Command Word	Binary display of input command register contents.				
								Memory Word	Binary display of word readout of C&S				
								Output Command Monitor	Individual command indicators.				

Figure 4.2-12: Computer And Sequencer Subsystem OSE Functional Description

1) Program Control--This block controls the selection and sequence of stimuli, measurement, display, and recording devices. Control information is supplied by the punched tape and includes the test, test limits, signal to be measured, and displays to be used.
2) Stimuli--Stimuli are routed by tape input under program control.
3) Measurement--An appropriate measurement device is selected by program control to measure the output being tested. Measurement devices are included to test serial pulse trains, serial and parallel digital data, and pulse characteristics.
4) Recording--Test identification and results are recorded on a printer for immediate use. Magnetic-tape recordings store data for further analysis.
5) Visual Displays--These are provided for use in manual operation. In the event of a failure, they provide the operator with data to diagnose and isolate faults.
6) Timing--Spacecraft time is measured and checked against the SS OSE internal reference. The timing circuits generate SS OSE internal clocking signals and include the capability of updating spacecraft time under program control.
7) Interface Circuits--Serial-to-parallel data conversions, test point selection, and continuous monitoring of computer and sequencer outputs are accomplished by the interface circuits under program control.

Test Sequence--A typical test sequence begins by storing a specific test subroutine in the C&S memory and verifying the results of this loading. Each memory location is loaded sequentially by a tape input word group specifying memory location and contents.

This information is assembled in the input registers and serially transmitted through a C&S input line in the form of a real-time command to store a word in C&S memory. Verification of storage proceeds in a similar manner by transmitting a real-time command to read out the same memory locations. Each memory location is then compared with the tape data in the input registers to verify storage. While the C&S is executing its subroutine, the output command lines are exercised by advancing the spacecraft clock to speed up the mission sequence. Mission time is also kept in the SS OSE clock for comparison. The C&S then initiates the time reference sequence. The program control circuits select the appropriate test point, specified by tape input, to detect the presence of a desired output. Other points are continuously monitored during a test sequence to detect the presence of an output on any line other than the one being tested.

This typical test sequence is preceded by tests to establish basic operational capability (i.e., acceptable voltage levels, pulse width, spacing, rise, and fall times) and verify "safe" status of the C&S. Pulse characteristics are measured by the time interval meter. An output data line is energized by program control, and pulse characteristics are measured by a time-interval meter and compared with programmed limits. Similarly, output levels measured by a digital voltmeter are compared with programmed limits. The SS OSE continuously monitors the C&S subsystem status line to halt or prevent the initiation of any test during a C&S "not ready" condition.

4.2.7.2 Interfaces

SS OSE with STC--Control of the computer and sequencer SS OSE in the STC is through the SS OSE tape reader control, which provides random access to tests. The STC provides 24-bit serial data with address and function coding to control the tape reader sequence. The control of the SS OSE equipment is contained in the program stored on the punched tape. All C&S outputs are monitored by the SS OSE and by the STC for simultaneous evaluation. Isolation buffers are included between the SS OSE and the STC. SS OSE stimuli are also supplied to the STC adapter for monitoring.

Hard-line data are evaluated simultaneously in the SS OSE and STC. Telemetry data are evaluated by the STC. A telemetry display is supplied by the STC for SS OSE operator information. The SS OSE supplies an alarm signal to the STC in the event of a malfunction or unacceptable measurement.

SS OSE with C&S--The SS OSE interface with the computer and sequencer subsystem is to mate with all operational inputs and outputs as shown in Volume A, Figure 4.1.9-10. When the SS OSE is used as part of the STC, the SS OSE interfaces through the test connector. In addition, a dummy load box in a separate unit simulates passive loading of the spacecraft interface cables when the C&S is disconnected during system test. This permits system testing to continue without C&S functions, while the C&S is checked out independently.

4.2.7.3 Performance Parameters

Performance characteristics of the computer and sequencer subsystem are shown in Table 4.2-7, along with the means of verifying these parameters by the SS OSE.

TABLE 4.2-7: PERFORMANCE CHARACTERISTICS

PARAMETER	SS OSE CAPABILITY
Memory - each (2) Capacity--1024 words	Memory can be loaded from the SS OSE input register by either tape input or manually inserted data. The contents of the C&S subsystem memory can be readout to the SS OSE register on command. The contents of the two registers in the SS OSE are automatically compared for verification of C&S memory contents.
Command Capability 150	All commands are monitored to detect the occurrence of an undesired command. The desired command is initiated by command from SS OSE, either direct or by advancing spacecraft time. Output command characteristics are also evaluated.
Bit Rate Operating 6.4-kc	The clock frequencies of the C&S subsystem are measured to verify the operation of the countdown chain. Reference frequency is provided by the SS OSE.
Word Length External - 28 bits Internal - 27 bits	Word length of the SS OSE registers are compatible with the C&S subsystem word length.
Program Routines Immediate Execution Modes (Highest Priority) Magnitude Computation, Comparison and Execution Discrete Execution Jump (Instruction Sequence Modification) Store Program Address Execution Telemeter Memory	Program routines are verified by simulating a complete mission sequence under control of the SS OSE.
Stored Program Mode Magnitude Computation, Comparison and Execution Discrete Execution Wait Time Computation, Comparison and Execution Jump (Instruction Sequence Modification) Compare Time Comparison and Execution	

TABLE 4.2-7 (Continued)

Interrupt Modes Store Sun Occultation Time Store Canopus Occultation Time Spacecraft Time Computation, Comparison and Execution Telemetry Programmer Data Terminator Crossing Time Inertial Reference	
Storage Modes Automatic Sequential Storage 1 Command Word/Stored Word Random-Access Storage 2 Command Words/Stored Word	
Computation Word Addition 1 Word/5 Msec Serial Comparison 26 Bits/5 Msec Magnitude Addition 10 Bits/Second	
Spacecraft Clock Type--Binary Capacity 776 Days, 17 Hours, 37 Minutes, 44 Seconds Least Significant Bit 1.0 Second Accuracy - 1 Part in 10^6 in 215 days Reset Modes - To Zero-Manual, Total Count, and Command Anytime - Command	The operation of the spacecraft clock is verified by comparison to the SS OSE clock. The SS OSE clock is run and advanced in synchronism with the spacecraft clock. The accuracy requirement of 1 part in 10^6 in 215 days is not verified in the C&S SS OSE since the spacecraft reference oscillator is not part of this subsystem. The reference for the C&S spacecraft clock is a 19.2-kc oscillator in the SS OSE during subsystem tests.
Velocity Conversion Register Range--200 pps Difference--25 parts in 200	The velocity conversion register is verified by simulated accelerometer pulses controlled by the SS OSE program. The programmed engine burn time and resultant commands are verified in the SS OSE.
Telemetry Memory Storage 16 (26-bit words/processor)	Telemetry memory storage is read out and compared to SS OSE programmed limits.

4.2.7.4 Physical Characteristics

The computer and sequencer subsystem OSE is housed in two standard NASA/JPL racks as used in Mariner C. The control and displays are grouped for operator convenience. The commercial equipment consists of an optical punched tape reader, time interval meter with counter, paper printer, digital voltmeter, and magnetic-tape recorder. The functional elements are as shown in Figure 4.2-12 . Primary power input is 115-volt, 60-cps, single phase of 1500-watts. Power for the computer and sequencer is supplied by the C&S SS OSE during subsystem tests.

4.2.7.5 Trade Studies

The design of the computer and sequencer subsystem OSE involved selection of a method for implementing the test program control. The two alternate methods considered were computer or punched tape reader. The computer offers greater speed and flexibility, but the tape reader proved adequate for this application and has sufficient design margin. The tape reader also offers less complex programming and lower cost. Accordingly, the tape reader program control was selected for computer and sequencer subsystem OSE.

4.2.8 Structural and Mechanical Subsystem OSE

OSE specifically applicable to structures such as alignment fixtures are identified at the system level in Section 3.3.5, and also serve as structures OSE. OSE required for mechanisms can be divided categorically into two groups: (1) that required for deployment devices, and (2) that required for separation devices. The first category includes a zero-g simulator for verifying that the several deployment devices will actuate, and a hydraulic-type test bench for checking the deployment actuators.

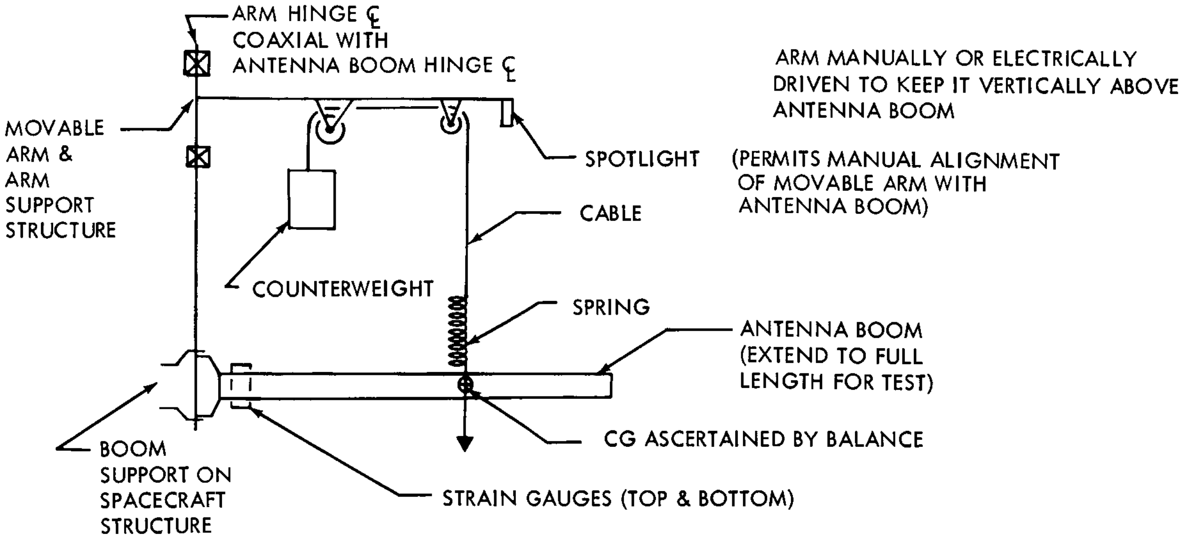
The same equipment can be used to conduct similar tests on GFE items such as the science boom and VHF relay antenna.

The second category includes equipment for installing and releasing the V-bands, which effect emergency separation of the Flight Capsule and operational separation of the Planetary Vehicle. This category includes: (1) strain-measuring equipment for tensioning the V-bands, (2) protective shields for personnel safety, and (3) firing devices for initiating V-band separation.

4.2.8.1 Zero-g Simulator (Figure 4.2-13)

This device creates, within reasonable limits, a zero-g condition while the low-gain antenna, medium-gain antenna, high-gain antenna, and solar panels are deployed and locked after being released from their stowed positions. The tests are conducted after the aforementioned items have been attached to the spacecraft. The device must accommodate angular movements of up to 150 degrees in approximately 10 seconds. Inertia and response characteristics of the OSE must not invalidate the test.

Functional Description -- The test equipment consists of suitable overhead counterbalance devices which create an essentially zero-g condition during the deployment cycle for the mechanism undergoing test. The support device tracks the mechanism during its deployment cycle to ensure that a valid test is obtained. The support elements are soft straps or cushioned pads to preclude damage to the deploying components. The only data required is the time required for deployment and locking. The test equipment must be able to handle items that are approximately 15 feet long and traverse 150 degrees during their deployment cycle.



PROCEDURE:

1) WITH BOOM LYING ON FLAT SURFACE, APPLY STRAIN GAUGES TO SIDES OF BOOM (BOOM IS ROTATED 90° FROM INSTALLED POSITION)
2) ZERO OUT STRAIN GAUGE ERRORS
3) BALANCE BOOM TO GET CG, THEN INSTALL BOOM
4) ATTACH SPRING AT BOOM CG
5) ADD TO COUNTERWEIGHT UNTIL STRAIN GAUGES READ "ZERO"
6) RELEASE PIN-PULLER TO PERMIT VINSON ACTUATOR TO DEPLOY BOOM
7) KEEP MOVABLE ARM VERTICALLY ABOVE ANTENNA BOOM DURING DEPLOYMENT BY KEEPING LIGHT BEAM CENTERED ON ANTENNA BOOM

Figure 4.2-13: Zero g Simulator

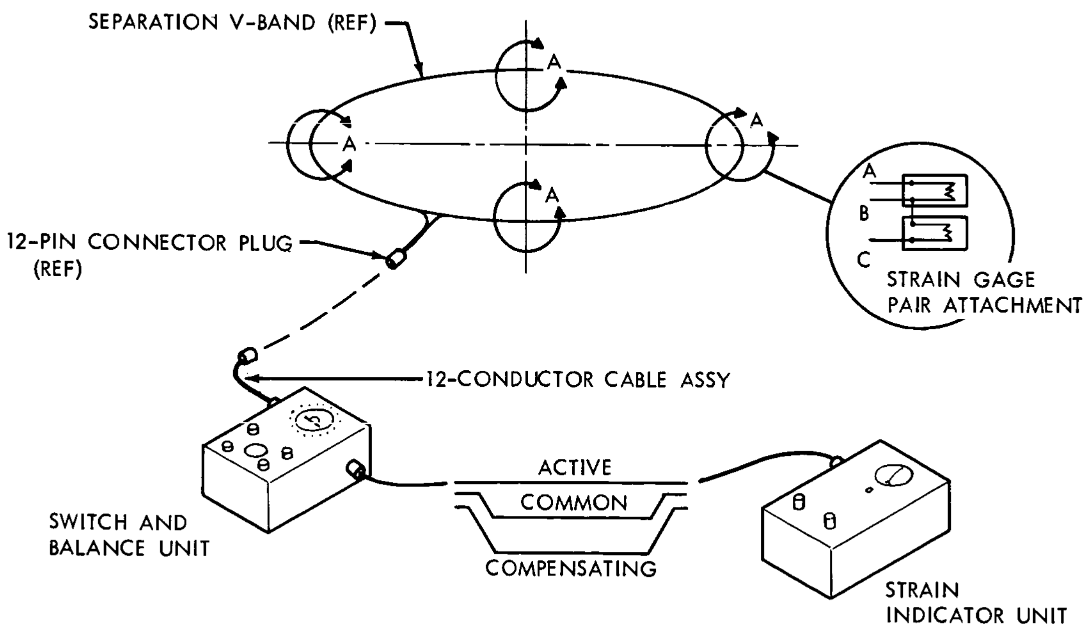


Figure 4.2-14: V-Band Installation Unit

Interfaces and Physical Characteristics--The only interfaces are attachment lugs, which might be integral with the deploying element. Approximate dimensions of the OSE are shown in Figure 4.2-13.

4.2.8.2 Vinson Actuator Test Bench

This OSE item is a hydraulic-bench-type device used to check the deployment time of the actuators used to deploy the low, medium and high-gain antennas and any GFE items that employ actuators for deployment. The test bench will measure the force exerted over the entire stroke of the actuator and the time required for extension when different loads are applied. Representative loads are 15 to 200 pounds during an actuator stroke of 6 to 12 inches.

Functional Description--The test bench provides clevis-type fittings for mounting the actuators, a means of compressing the actuator against its internal coil spring, and a means of maintaining a fixed load on the actuator as it extends. A hydraulic circuit containing a variable-pressure regulator is selected because of the ready availability of such equipment. Pressure-relief valves incorporated in the hydraulic circuit ensure the safety of operating personnel.

Interfaces and Physical Characteristics--The only interfaces required are the clevis attachments required to mount the actuators. The physical size and weight are comparable to those of a regular hydraulics test bench.

4.2.8.3 V-Band Installation Unit

An active tension-measuring device is required to ensure that equal tension loads are achieved when tightening the tie bolts in the emergency capsule and Planetary Vehicle separation bands. This device consists of a conventional switch and balance unit coupled to a strain indicator unit of the types employed in strain-gage technology. The same units are used for both V-bands. Continuous calibration is required during the testing operation.

Functional Description--A functional flow diagram of the equipment set-up is shown in Figure 4.2-14. The switch and balance unit and the strain indicator are connected to a pair of strain gages mounted on each end of the four segments of the V-band. Readings from each strain-gage pair are taken in sequence as the band is tightened to the required load. Before each reading, a comparison check is made against a calibration unit to prevent overtensioning the band. Data are manually recorded from the instrument readings.

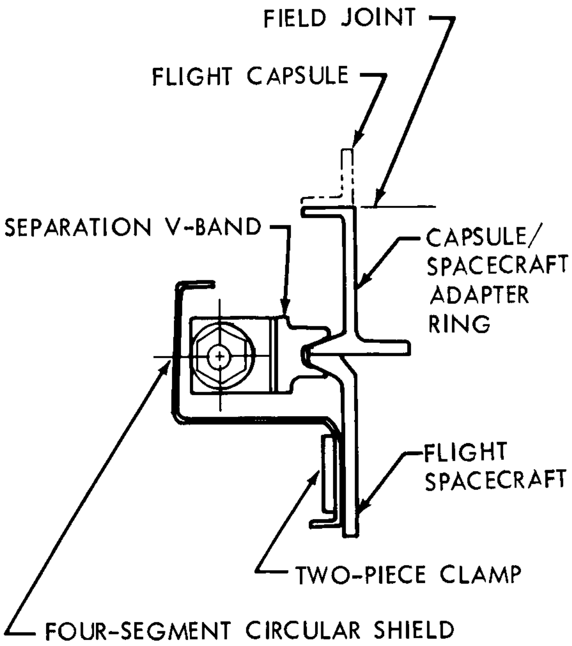
Interfaces--The only interface required is that with the connectors in the strain-gage circuitry. For ease of attachment, all strain-gage leads terminate in a common connector. Power for operating the equipment is self-contained within the OSE.

4.2.8.4 Protective Shields

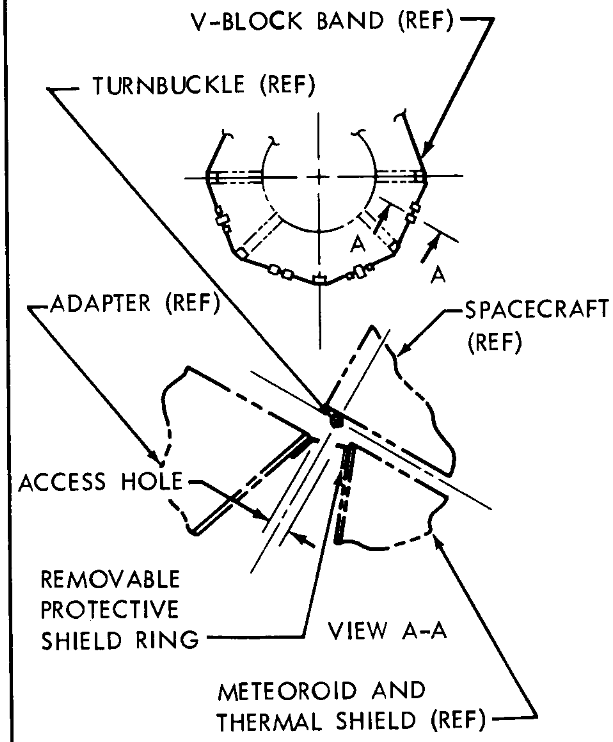
Shields are provided to protect personnel should either V-band fail. Since one V-band mounts on a circular ring, while the other mounts at eight points, two different shields are required. Both are made up of segments to facilitate installation and incorporate wrenching access holes for tightening the V-band turnbuckles. Their concepts are shown in Figure 4.2-15.

4.2.8.5 V-Band Separation Initiating Device

To give maximum assurance that the V-bands will separate, it is proposed



FLIGHT CAPSULE
EMERGENCY SEPERATION



PLANETARY VEHICLE ADAPTER

Figure 4.2-15: Protective Shields

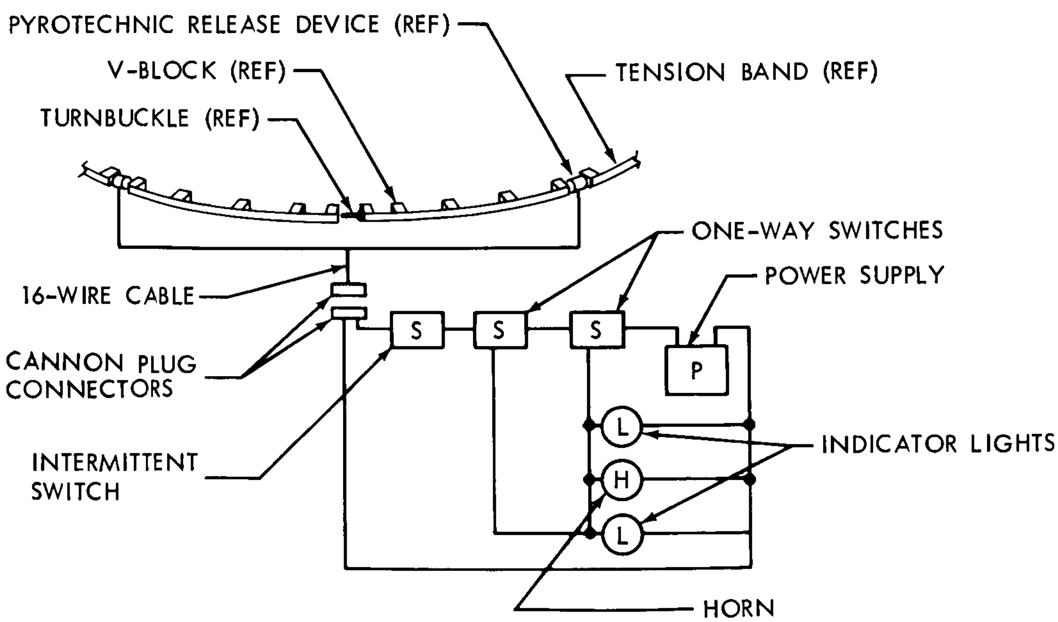


Figure 4.2-16: V-Band Separation Initiating Device

that each V-band be assembled and fired one time under conditions of maximum and minimum tensioned loads. The band would then be reinstalled and torqued to the midrange of the maximum and minimum values. Replacement of the V-band would require that the test be repeated. A pyrotechnic firing circuit with a self-contained power supply is required to perform the test. Also required are the OSE items described in Sections 4.2.8.3 and 4.2.8.4. The device consists basically of a 6-8 volt dc power source contained in a suitable box, which also houses a voltmeter and ammeter and provides a mounting platform for a horn, two guarded switches, and a submerged push-button switch. An outgoing wire bundle containing 16 wires and terminating in a 16-pin connector connects the device to the V-band-release pyrotechnic circuit.

Functional Description--The unit is shown schematically in Figure 4.2-16. It contains its own power supply to minimize inadvertent firing opportunities, supplying 4.5 to 6 amperes to the V-band pyrotechnic circuit. Firing switches are spring-loaded in the off position and covered by spring-loaded guards. These switches, plus a submerged (below the surface) push-button switch, are in series in the firing circuit to prevent inadvertent firing. The closure of either of the protected toggle switches causes the warning horn to sound. The device is so wired that the outgoing signal can be initiated only after the horn circuit relay has been closed. The upper face of the housing contains a painted circuit diagram with red warning lights to indicate when a circuit is "hot."

Interfaces and Physical Characteristics--The only interface is with the V-band adapter connector plug. This connection is required only during the test period. Dimensionally, the unit is housed in a box approximately 10 inches on each side.

4.2.9 Pyrotechnic Subsystem OSE

The pyrotechnic subsystem OSE is designed to automatically test the integrity of the pyrotechnic subsystem. This includes verification of firing signals on selected output lines and detection of undesired signals on the remaining output lines. Tests verify the redundant firing circuits by exercising each redundant input line separately, then both simultaneously. This corresponds to a single failure on each input line followed by a normal firing mode. Safe/arm status signals from the pyrotechnic arming switch (PAS) and the separation initiation timer (SIT) preclude starting any test unless the PAS or the SIT arm the pyrotechnic subsystem. The PAS, SIT, and electroexplosive device (EED) units are tested separately. For system tests, the subsystem OSE cannot initiate any firing commands unless specifically directed by the STC. The subsystem OSE acts as a monitor, controlled by the STC, to indicate safe/arm status and as an output signal detector.

The EED's for both the spacecraft and adapter are checked by a portable manual tester. The tester measures continuity and resistance of the EED's. Positive current limiting is provided. This tester also checks for the presence of voltage on the cables prior to connector mating.

4.2.9.1 Functional Description

The pyrotechnic subsystem OSE functions are listed below and are shown in Figure 4.2-17. The functions of each block are described in Figure 4.2-18. A typical test sequence is presented to demonstrate the re-

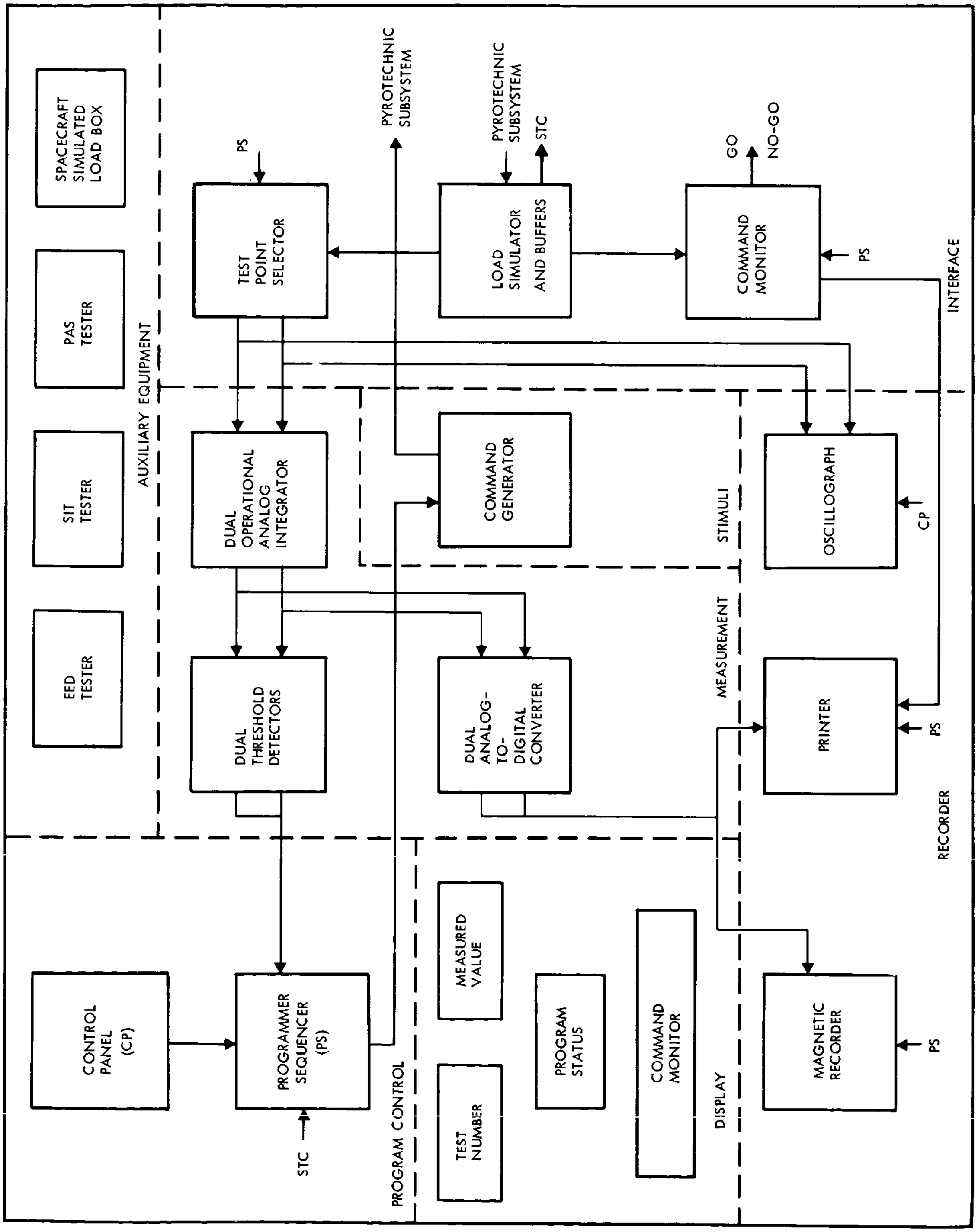


Figure 4.2-17: Pyrotechnic Subsystem OSE

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PROGRAM CONTROL SECTION		STIMULI SECTION		DISPLAY SECTION	
Programmer Sequencer	The sequencer switch controls the program through a diode matrix programmable pin board on one minute intervals. The functions performed by the programmer are: 1. Select and initiate commands to the pyrotechnic subsystem. 2. Select test point for measurement. 3. Provides digital test number to the display printer, and magnetic recorder. 4. Provides control to the printer and recorder. 5. Provides inhibit signal to command monitor for the selected output drivers.	Command Generator	The command generator provides gated pulse trains to the pyrotechnic drivers. The normal input from both C&S and command subsystems are verified in separate tests. The command generator is inhibited when the SSOSE is used in the system test complex unless specifically directed to issue a firing command by the STC.		Numerous displays are provided to aid the operator in monitoring and controlling the test set. These displays are:
				Test Number	Decimal display of test being performed.
				Measured Value	Decimal display of the measured value. Two displays are required for channels "A" and "B".
				Command Monitor	Individual indicators showing the firing of a driver. These indicators light only when an undesired firing occurs. Desired firings are inhibited by the program.
				Status: Go/No-Go	Indicator showing acceptance or rejection of the measured value.
Control Panel	The control panel provides the capability of manual intervention in the programmed sequence. The controls are: 1. Automatic--the sequencer switch steps through the program at one minute intervals. 2. Manual--the selector switches provide random access to command any pyrotechnic driver. An initiation switch is used to send selected command. 3. Remote--control of programmable matrix is through the STC. The control to the stimuli section is inhibited when commands are initiated through the C&S or command subsystem in systems test. 4. Arming switch--Provides arming command to pyrotechnics subsystem. 5. Separation initiation--provides a simulated separation initiation timer sequence			Ready-Hold	Indicator showing pyrotechnic drivers are ready to fire or waiting for capacitor recharge.
				Safe/Arm	Indicator showing pyrotechnic subsystem is armed.

1

MEASUREMENT SECTION		RECORDING SECTION		INTERFACE SECTION		AUXILIARY EQUIPMENT	
bera- ional halog hte- rator Dual)	The measurements is to determine the ampere milli-seconds above the threshold firing current. Two channels are provided to permit the measurement of two drivers fired simultaneously. The selection of inputs is accomplished by the test point selector. The output of the integrator is fed to the threshold detector and the analog-to-digital converter.	Printer	Provides a printed paper record of test number, measurement value, malfunctioning driver number and no-go indication for local test history. Selection is available for printing all data or only no-go data. The printer is controlled by the programmer.	Simulated Load	The remote loads for the pyrotechnic drivers are inert squibs to simulate the normal load. A current transformer is included for RC discharge current measurement.	EED Tester	Remote test set for measuring resistance and continuity of electroexplosive devices. Provisions for positive current limiting is included.
ireshold tector ual)	Determines acceptability of the driver outputs.	Magnetic Recorder	Provides a magnetic tape record of test number and measured value for test data comparison and trend analysis. The recorder is controlled by the program.	Test Point Selection	The driver output to be measured is selected by a TPS controlled from the program. Two outputs can be selected simultaneously.	SIT Tester	Separate tester for measuring timed switch closure of separation initiation timer.
ialog- ı-Dig- ial nver- or	Provides quantitative decimal value of measured output for display and recording.	Oscillograph	Provides analog recording of firing circuit RC discharge. Control is manual with input being selected by the test point selector.	Command Monitor	The command monitor provides a continuous monitoring of all driver outputs to detect an inadvertent firing. The indicator for the desired firing is inhibited by the program. If a firing occurs, a stop command is issued, the indicator on the display panel lights for the specific malfunctioned line, and the identity of the fault is recorded.	PAS Tester	Separate tester for measuring switch closure of pyrotechnic arming switch.
						Dummy Loads	Provides simulated loads for other spacecraft subsystems when the pyrotechnic subsystem is disconnected for independent tests.

Figure 4.2-18: Pyrotechnic Subsystem OSE Functional Description

2

lationship of the functional elements.

Major Elements--The pyrotechnic subsystem OSE uses seven functional elements.

1) Program Control--Provides the control of command initiation and pyrotechnic output evaluation.
2) Display--Provides visual display of test set operation and results.
3) Stimuli--Provides command generation to the pyrotechnic subsystem.
4) Measurement--Provides evaluation of pyrotechnic driver outputs.
5) Interface--Provides continuous monitoring, loading, and test point selection.
6) Recorder--Provides permanent record of test results.
7) Auxiliary Equipment
   a) EED Tester--Portable tester for continuity and resistance measurements of electroexplosive devices.
   b) SIT Tester--A separate tester measuring the time switch closure sequence of the separation initiation timer.
   c) PAS Tester--A separate tester for measuring the switch closure of the pyrotechnic arming switch.

Test Sequence--The test procedure begins by connecting simulated pyrotechnic devices to the firing lines and arming the pyrotechnic power switching unit (PPS) by actuating the SIT, PAS, or bypass power switches through the SS OSE. Automatic testing starts by actuating the programmer sequencer from the control panel. Ampere-seconds delivered to the simulated pyrotechnic devices are measured and compared with a known standard. An acceptance signal causes the program control circuits to proceed to the next test. Display devices monitor program status and test results.

Records of test results are available in printed form for immediate use and on magnetic tape for subsequent data analysis. Signals from the stimuli generator are selected by the program control circuitry to energize input lines from the associated subsystems i.e., command or computer and sequencer subsystems. All PPS driver lines are monitored during testing to provide an indication of undesired signals on all except the desired output line. The desired output line indicator is inhibited by the programmer sequencer.

4.2.9.2 Interfaces

SS OSE with STC--The commands from the STC to the pyrotechnic subsystem OSE is a 24-bit serial data train with address and function coding. The STC controls the operation of the SS OSE through the programming matrix. All pyrotechnic hardline outputs are monitored by the SS OSE and the STC for simultaneous evaluation. Buffers are included in the SS OSE for isolation to the pyrotechnic subsystem and STC. Telemetry data are evaluated by the STC and a telemetry display is supplied by the STC for SS OSE operator information. The SS OSE supplies an alarm signal to the STC in the event of a malfunction or unacceptable measurement. SS OSE stimuli are available to the STC for monitoring; however, the stimuli are not supplied to the pyrotechnic subsystem unless specifically commanded by the STC.

SS OSE-SS--The SS OSE interfaces with the pyrotechnic subsystem to mate with all operational inputs and outputs as shown in Volume A, Figure 4.1.11-1.

Since the SS OSE load simulators are remote from the main test set, independent lines for each pyrotechnic output are connected from the loads to the main test set. A dummy load box is provided for spacecraft cabling when the pyrotechnic subsystem cables are disconnected in the STC. This permits simulated passive loading for other subsystems while the pyrotechnic subsystem is being tested independently.

4.2.9.3 Performance Parameters

The performance characteristics of the pyrotechnic subsystem are shown below, along with the means of verification of the parameters by the SS OSE.

Input command signals are 4.8. kc gated on for 100 milliseconds. These signals are provided by the SS OSE stimuli generator on command from the programmer. Marginal amplitude and duration signal conditions can be simulated.

Output pulses to the electroexplosive devices are measured for energy above threshold, compared to a limit value and digitized for recording and display. All lines are monitored to detect any firings that were not commanded.

4.2.9.4 Physical Characteristics

The pyrotechnic subsystem OSE is housed in one standard NASA/JPL rack as used in Mariner C. The controls and displays are grouped for operator convenience. Commercial equipment consists of digital voltmeters, paper printer, and magnetic recorder. The test sets for the EED, PAS, and SIT are separate manual testers. Power input is 115-volt 60-cps, single phase, 1000 watts. Power for the pyrotechnic subsystem is supplied by the SS OSE during subsystem test.

4.2.9.5 Trade Studies

The design of the pyrotechnic SS OSE involved the consideration of alternate methods of implementation. The major items considered were the implementation of program control and automatic measurement of pyrotechnic driver output.

The program control is a choice between a tape reader program and a fixed sequencer program. The additional cost and complexity of the tape programmer is not justifiable since the test sequence is repetitive on multiple channels and testing speed is limited by the pyrotechnic subsystem. The fixed sequencer programmer was selected.

The automatic measurement of the pyrotechnic subsystem driver outputs is a choice between analog and digital integration of the ampere-milliseconds in the RC discharge above minimum firing current. The analog integration involves an operational analog integrator followed by a threshold detector for a check on acceptability and an analog-to-digital converter for display and recording. The digital integrator involves a voltage-to-frequency converter followed by an accumulation counter, gated by the threshold crossing of the current analog. A digital comparator is required to check for acceptability. The analog integrator is selected due to ease of implementation and lower cost.

4.2.10 Temperature Control Subsystem OSE

Passive temperature control subsystems cannot be checked out as a subsystem other than validating the optical properties of the coatings and radiative insulation and adjustment verification of the active elements, which are the louvers and backup heaters and temperature switches. The total system is validated by the space chamber test.

4.2.10.1 Functional Descriptions

The temperature control subsystem OSE is used identically at all locations of spacecraft testing and consists of the following:

1) Louver adjustment and calibration test set;
2) Heater continuity and burn-in test set;
3) Temperature switch calibration test set;
4) Thermal coatings batch validation (αs, E_{IR});
5) Insulation thermal conductivity validation;
6) Insulation emissivity validation.

Figure 4.2-19 presents the overall usage of the above equipment.

Louver Adjustment and Calibration--A test set is required for the determination of correct and accurate adjustment and calibration of each louver array. The blades are factory checked, checked before space chamber testing, and rechecked prior to installation at ETR. A schematic of the test set is shown in Figure 4.2-20.

Heater Continuity and Burn-In Test Set--The backup heater circuitry requires a periodic continuity check and an initial burn-in test. The test set is composed of a dc 0 to 100 volt supply and an ammeter arrangement as shown in Figure 4.2-21.

Temperature Switch Calibration Test Set--The make-contact and break-contact temperatures of the temperature switches are measured with the test set shown in Figure 4.2-22. The switches are activated by spot heating, and an ohmmeter circuit checks continuity.

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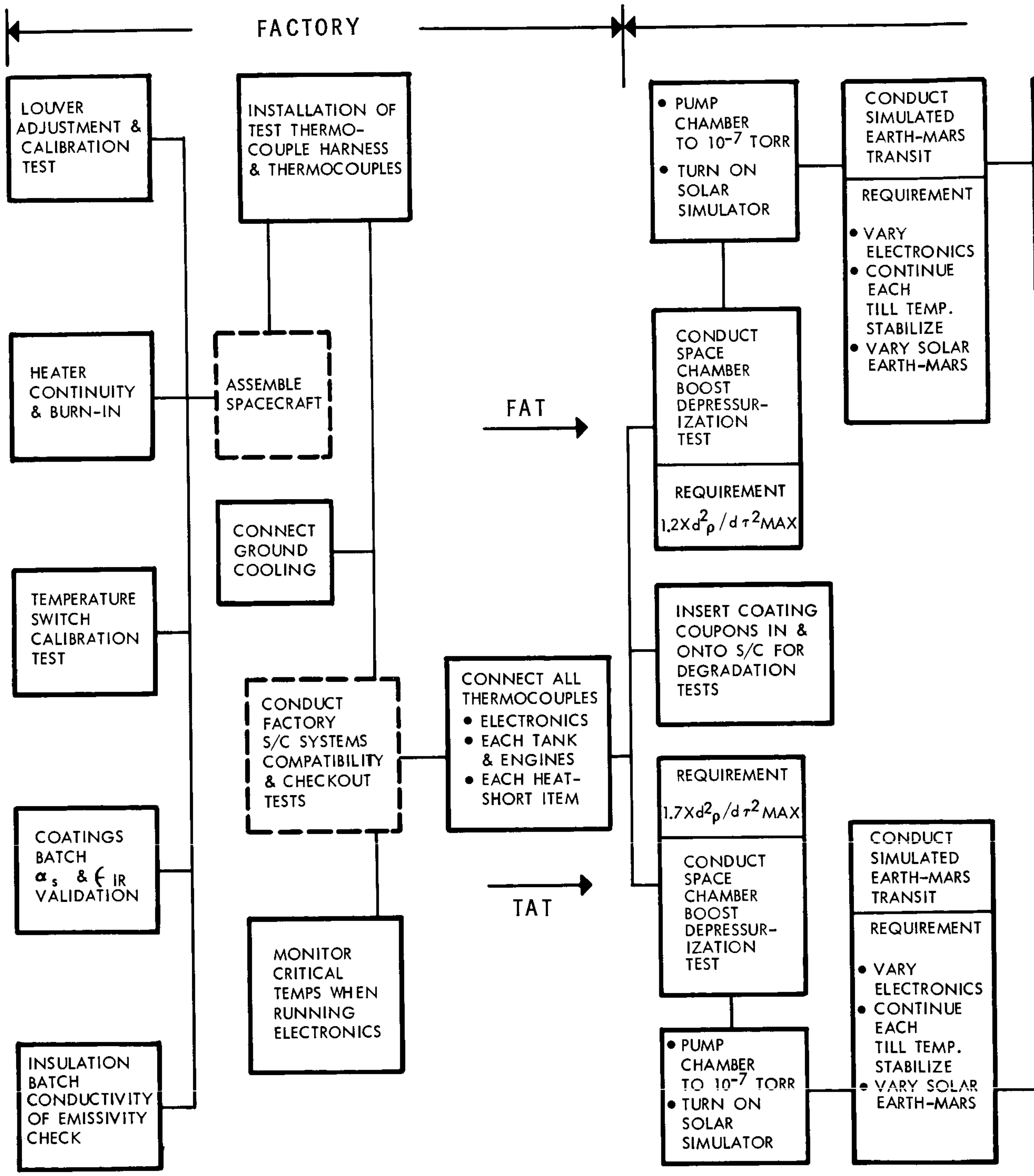
FACTORY
LOUVER ADJUSTMENT & CALIBRATION TEST
INSTALLATION OF TEST THERMO-COUPLE HARNESS & THERMOCOUPLES
HEATER CONTINUITY & BURN-IN
ASSEMBLE SPACECRAFT
CONNECT GROUND COOLING
TEMPERATURE SWITCH CALIBRATION TEST
CONDUCT FACTORY S/C SYSTEMS COMPATIBILITY & CHECKOUT TESTS
CONNECT ALL THERMOCOUPLES
• ELECTRONICS
• EACH TANK & ENGINES
• EACH HEAT-SHORT ITEM
COATINGS BATCH α_s & ϵ_{IR} VALIDATION
MONITOR CRITICAL TEMPS WHEN RUNNING ELECTRONICS
INSULATION BATCH CONDUCTIVITY OF EMISSIVITY CHECK
FAT
TAT
• PUMP CHAMBER TO 10^{-7} TORR
• TURN ON SOLAR SIMULATOR
CONDUCT SIMULATED EARTH-MARS TRANSIT
REQUIREMENT
• VARY ELECTRONICS
• CONTINUE EACH TILL TEMP. STABILIZE
• VARY SOLAR EARTH-MARS
CONDUCT SPACE CHAMBER BOOST DEPRESSUR-IZATION TEST
REQUIREMENT
$1.2Xd^2p/d\tau^2MAX$
INSERT COATING COUPONS IN & ONTO S/C FOR DEGRADATION TESTS
REQUIREMENT
$1.7Xd^2p/d\tau^2MAX$
CONDUCT SPACE CHAMBER BOOST DEPRESSUR-IZATION TEST
• PUMP CHAMBER TO 10^{-7} TORR
• TURN ON SOLAR SIMULATOR
CONDUCT SIMULATED EARTH-MARS TRANSIT
REQUIREMENT
• VARY ELECTRONICS
• CONTINUE EACH TILL TEMP. STABILIZE
• VARY SOLAR EARTH-MARS

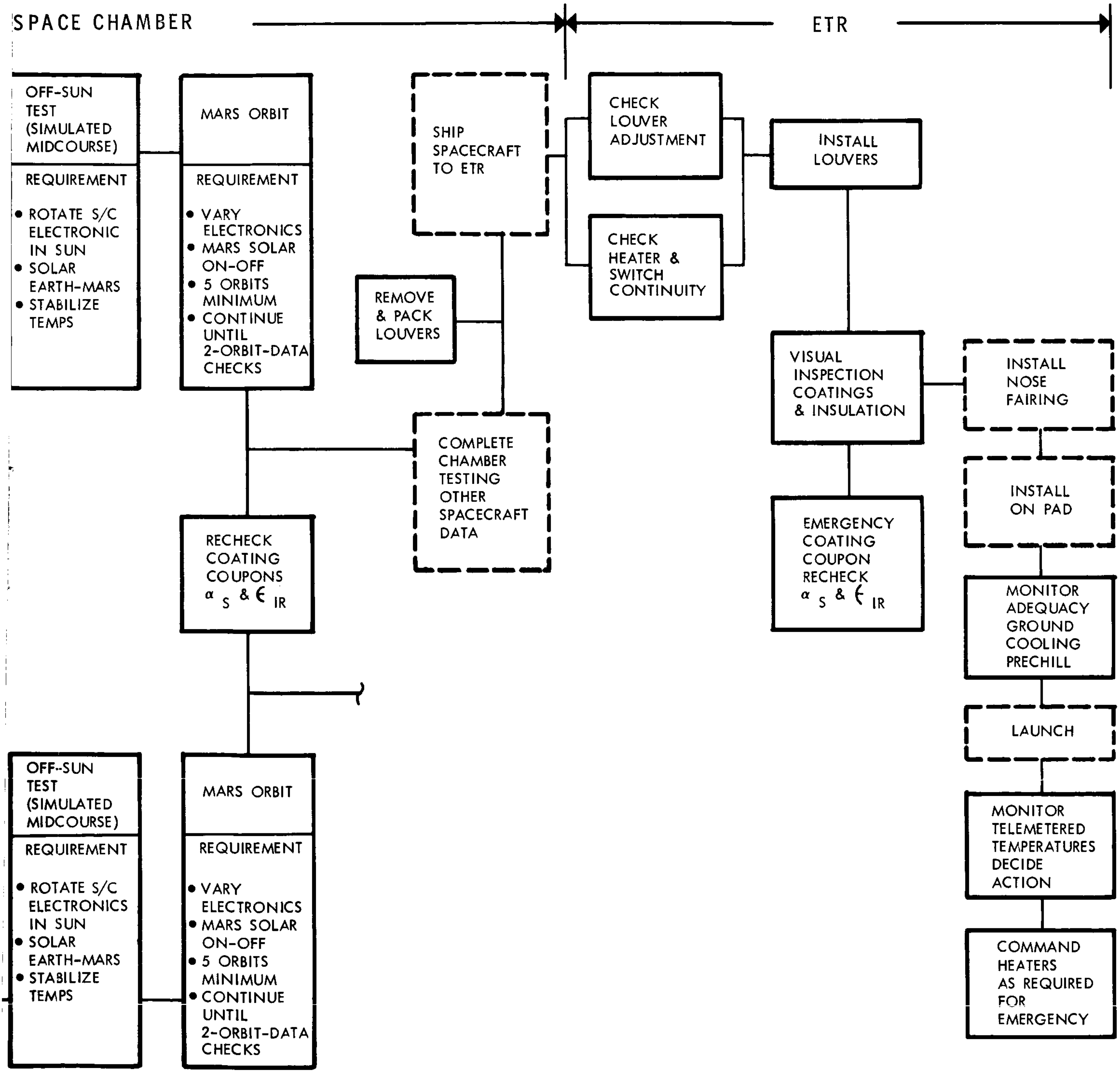

Figure 4.2-19: Temperature Control — OSE Functional Flow

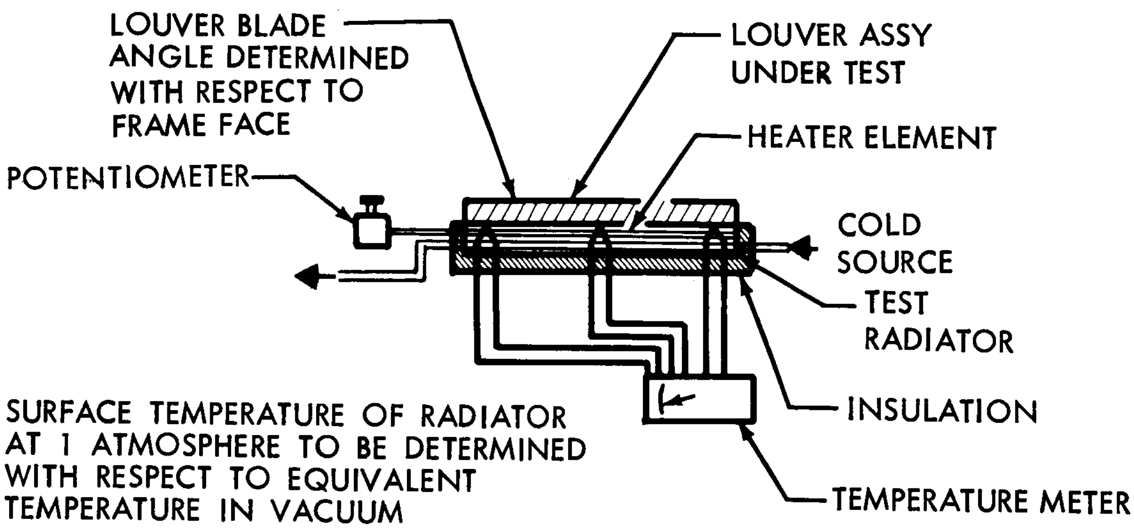

Figure 4.2-20: Test Set — Louver Adjustment And Calibration

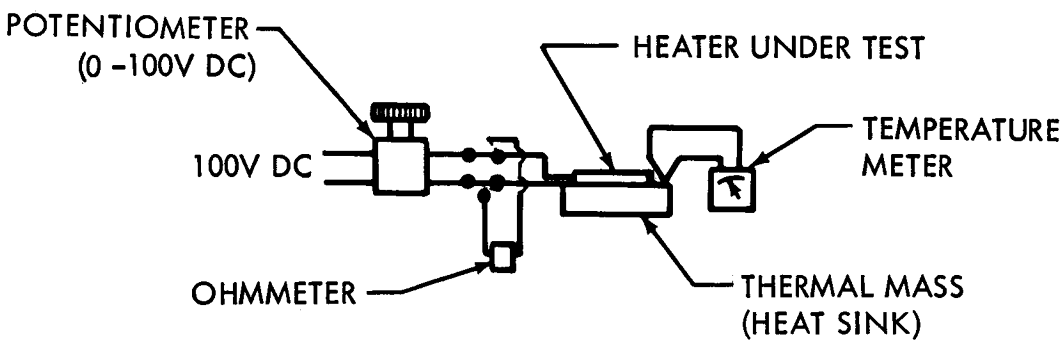

Figure 4.2-21: Test Set — Heater Continuity And Burn-in

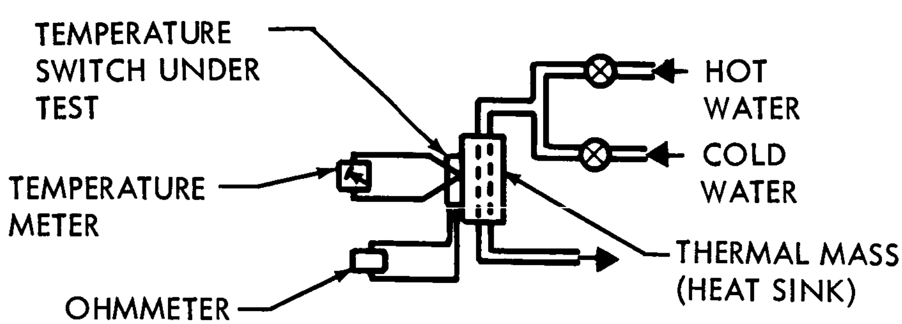

Figure 4.2-22: Test Set — Temperature Switch Calibration

Thermal Coatings Batch Validation--Absorptivity and emissivity are measured on individual thermal coating coupons. Measurements are required on each coating batch mixed, before and after space simulation testing, and as an emergency measure just prior to insertion of the spacecraft into the nose fairing. Specific measurement procedural control is maintained such that spectral measurements are integrated over a specified wavelength and later coupon evaluations are conducted using the same optical instrument or suitable known adjustment standards.

Instrumentation used for this type of measurement consists of:

1) Solar (0.25 to 2.6 microns)
   a) Gier-Dunkle directional reflectometer; or
   b) Beckman DK-2A spectroreflectometer.
2) Infrared (1.0 to 32 microns)
   a) Beckman IR-4 spectrophotometer with Hohlraum cavity.

Insulation Thermal Conductivity Validation--A recognized standard means of measuring thermal conductivities of bulk-type insulations is required on receipt and prior to use of Voyager materials. This is required to determine acceptable "Fiberglass" Scrim material and to measure the equivalent thermal conductivity of the finished aluminized mylar insulation blanket.

Insulation Emissivity Validation--To ensure uncontaminated aluminized mylar-type insulation blankets in the field, coupon samples of the material will accompany the spacecraft at all times. If contamination is suspected, it will be necessary to recheck the emissivity of a representative sample of this material. The Beckman IR-4 mentioned above can be used.

4.2.10.2 Interfaces

The thermal subsystem OSE will interface with the power and command subsystem OSE to accomplish intersubsystem compatibility during the heater and temperature switch continuity testing mentioned in Section 4.2.10.1.

In addition, thermal coating coupon samples, representative of those used on all other spacecraft subsystem surfaces, require validation as described in Section 4.2.10.1 above for the surfaces that are temperature critical.

4.3 PROPULSION SUBSYSTEM OSE

The Voyager Spacecraft Propulsion Subsystem OSE is the supporting test and servicing equipment that verifies the integrity and proper operation of the subsystem and provides gas and liquid servicing from fabrication and assembly through prelaunch operations.

4.3.1 Functional Description

The propulsion subsystem OSE consists of five major items whose functions are described individually in the following paragraphs.

4.3.1.1 Fuel Servicing Unit

The fuel servicing unit is a four-wheel trailer containing a control panel for propellant transfer, fume scrubbers for safe venting of fuel vapor, a fuel storage tank, and platform scales for precision weighing. Figure 4.3-1 is a schematic of the unit.

The fuel servicing unit is capable of loading one spacecraft at a time with the correct amount of clean, sterile propellant at a controlled rate. The control panel for fuel transfer is mounted at the rear of the trailer.

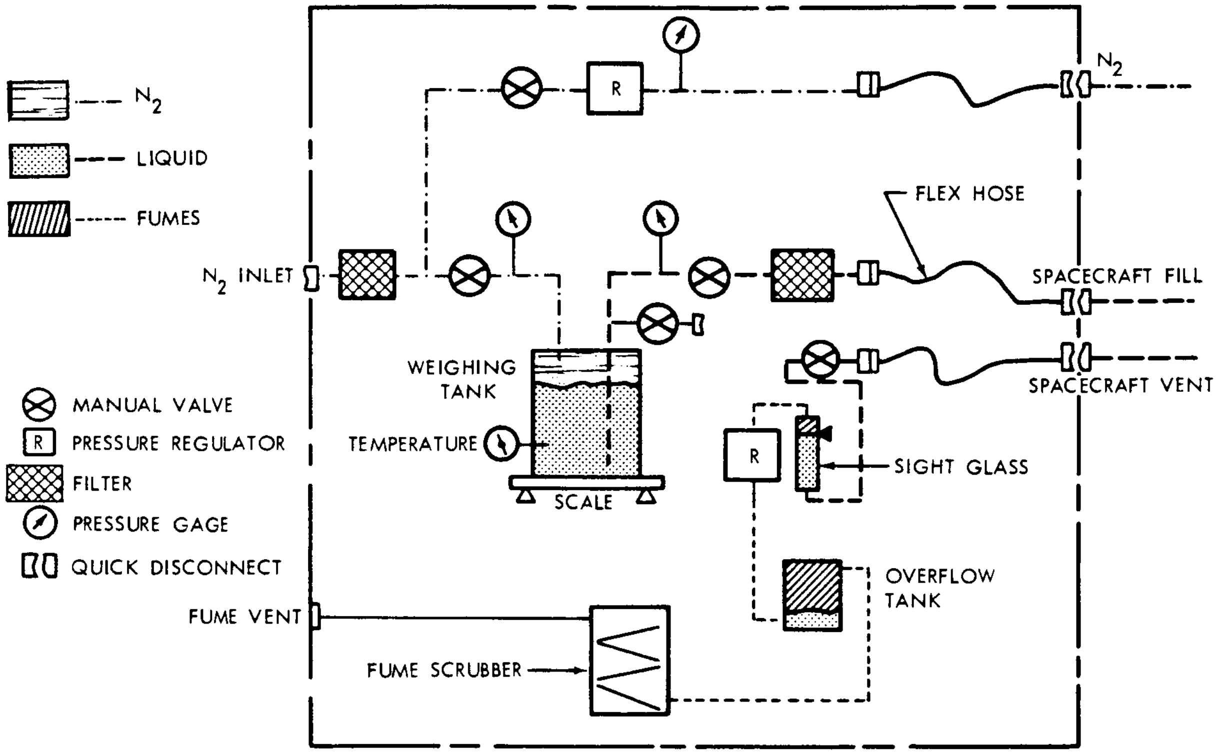

Figure 4.3-1: Fuel Servicing Unit

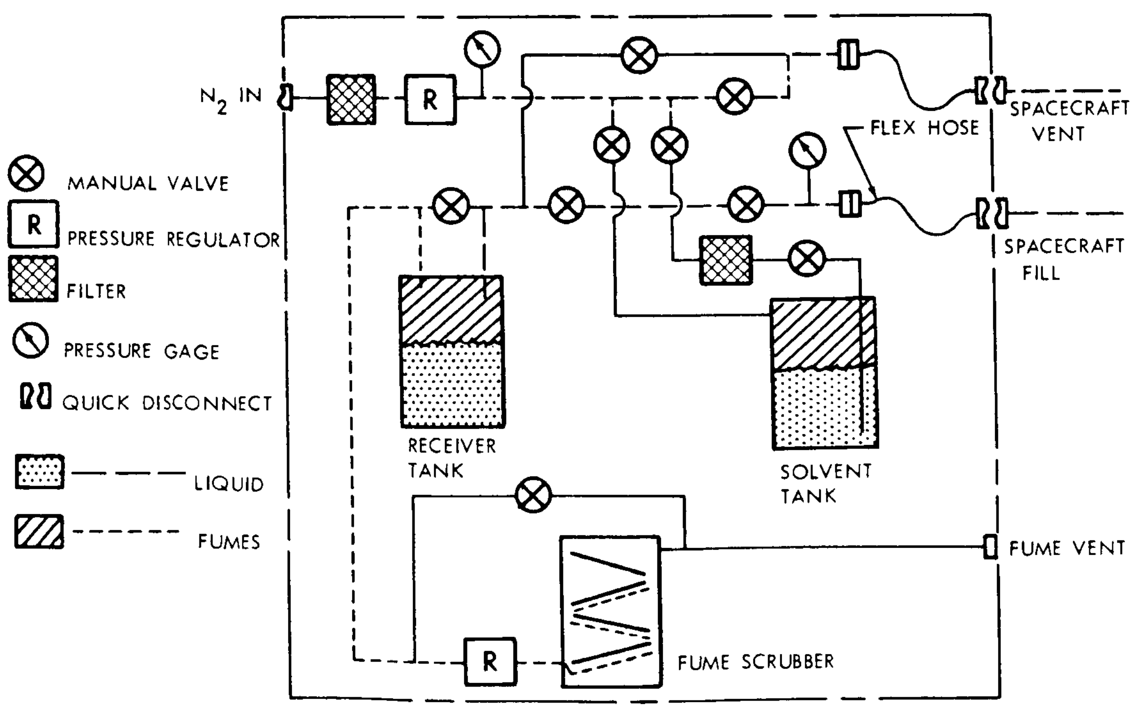

Figure 4.3-2: Propellant Decontamination Unit

The capacity of the fuel storage tank is 3100 pounds. A thermometer mounted on the tank indicates the temperature of the fuel. Propellant transfer from the tank is accomplished by nitrogen pressure. Before entering the spacecraft, the fuel passes through 5-micron filters. The amount of fuel transferred to the spacecraft can be determined within 0.10 percent with the platform scales. Fumes vented from the spacecraft pass through an acetic acid fume scrubber to reduce the hydrazine concentration to less than 5 parts per million.

Similar equipment has been built for the Surveyor program by Garrett Corp. and for the Lunar Orbiter program by the Standard Manufacturing Co. Techniques for making sterile connections for propellant loading will be studied and tested on the engineering test models.

4.3.1.2 Propellant Decontamination Unit

The propellant decontamination unit is mounted on a trailer. This mobile unit incorporates an isopropanol supply tank, receiving tankage for flushing solvent, flow control, monitoring equipment, fume scrubber, and a nitrogen pressurization system for liquid transfer.

The unit flushes, purges, and dries the spacecraft propellant subsystem and the fuel servicing unit. Figure 4.3-2 is a schematic of the unit.

The unit operates in a closed liquid-vapor loop with the spacecraft propulsion system or propellant servicing unit and controls toxic vapors the same as the servicing unit. It is functionally divided into a 0 to 100 psig nitrogen circuit, a solvent supply circuit, a solvent/propellant receiving circuit, and a vent and vapor disposal system.

The decontamination unit supplies, circulates, and retains multiple solvent flushes of the spacecraft propellant system and the fuel servicing unit. It detects concentrations of N_2H_4 in solvent in 10-ppm increments over a range of 0 to 100 ppm. In addition, it dilutes hydrazine vapors to less than 5 ppm.

4.3.1.3 Nitrogen Servicing Unit

The nitrogen servicing unit is a rack-mounted vertical control panel that contains regulators, valves, and interconnecting plumbing necessary to control the flow and pressure of a nitrogen and helium source. The base of the rack supports the filter and dryer units. Figure 4.3-3 shows a schematic of this unit.

This unit performs the functions required to regulate and distribute 6000-psig nitrogen, 2200-psig nitrogen and 2200-psig helium, for use in the following tasks:

1) To provide the pressurized nitrogen and helium for the propulsion and reaction control systems test unit;
2) To provide the nitrogen charge (0 to 100 psig) for the propulsion system purge, dry, and flush unit;
3) To provide the nitrogen charge (0 to 100 psig) for the fuel servicing unit;
4) To provide the nitrogen charge (0 to 100 psig) for the TVC fluid servicing unit.

The unit is trailer mounted. All gas delivered from the unit is dried to -100^{o}F and filtered to 5 micron nominal.

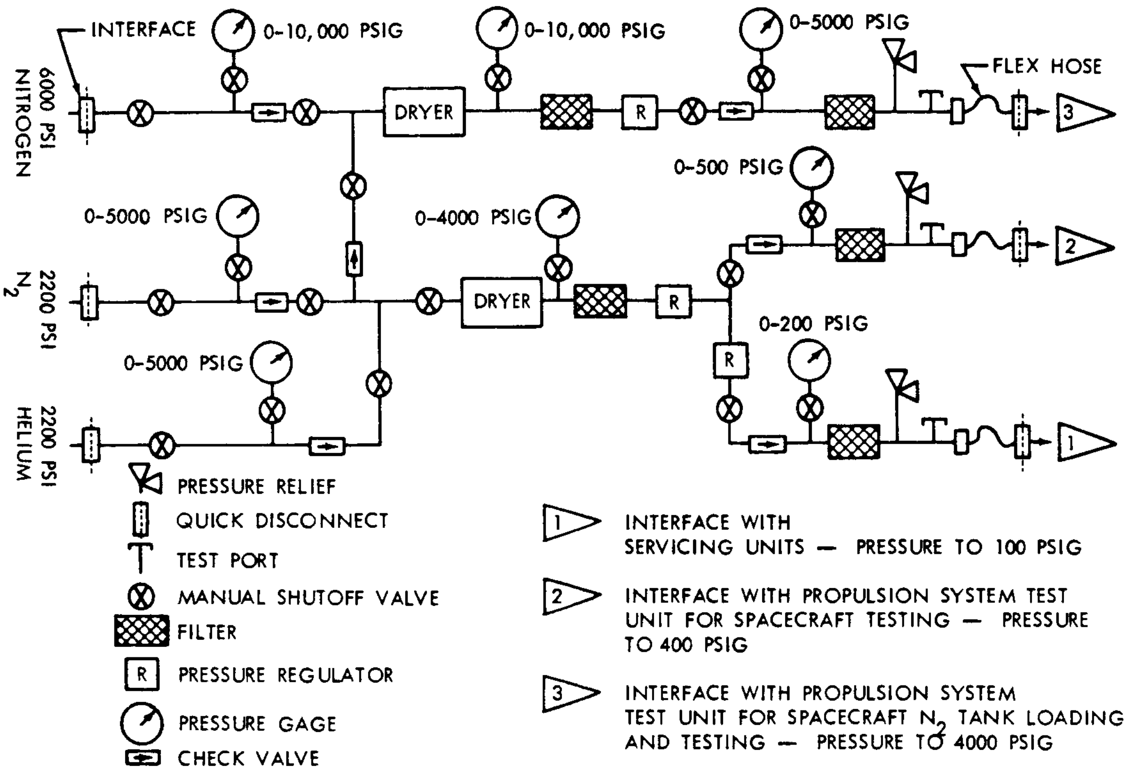

Figure 4.3-3: Nitrogen Servicing Unit

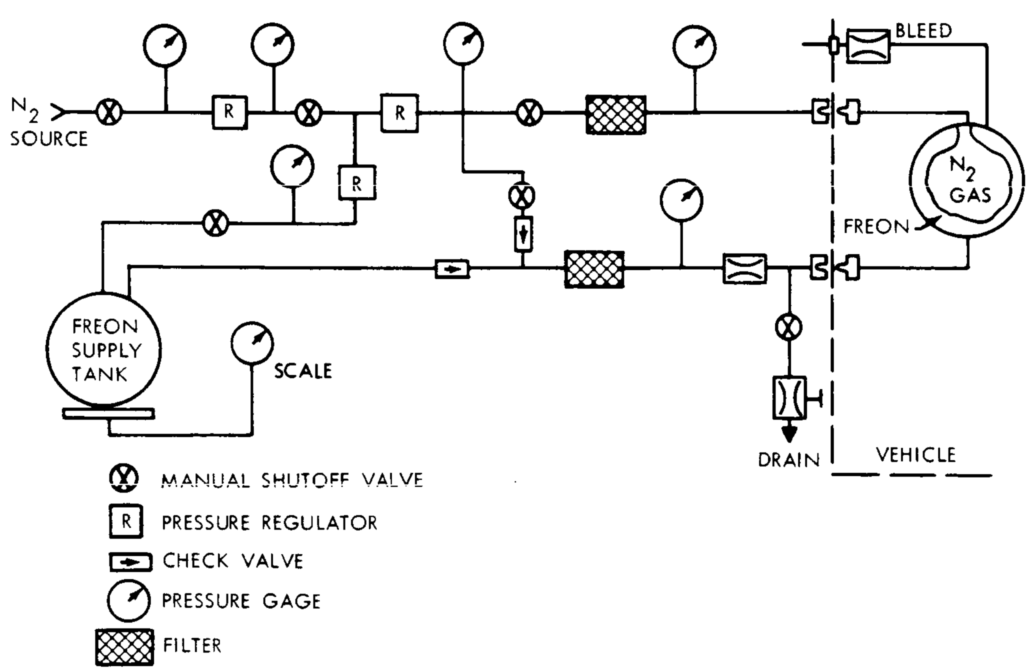

Figure 4.3-4: Freon Servicing Unit

4.3.1.4 Propulsion and Reaction Control Test Set

The set performs tests on the spacecraft propulsion, thrust-vector-control, and reaction-control systems. Travel and angle of mechanical movements are measured versus electrical signal input. Other tests include leak checks and actuation and response testing on the electrical solenoids in the propellant and nitrogen pressure circuits of the spacecraft midcourse and orbit-trim propulsion, orbit-insertion propulsion, thrust-vector-control, and reaction-control subsystems. The electrical test equipment checks the response of the reaction control nozzle valves and midcourse propulsion engine jet vanes to simulated autopilot signals. It also tests the orbit-insertion-motor TVC system response to simulated autopilot signals. The test set also: (1) measures regulated pressures to 264 $\pm$ 2 psig in the midcourse propulsion system; (2) measures regulated pressures to 50 psia in the reaction-control system; (3) detects leaks to 1×10^{-7} atm cc/sec; (4) filters gases just prior to spacecraft entry to 0.3-micron nominal, 1.2-micron absolute; (5) verifies dryness over dewpoint range of 21°C to -62°C (70°F to -80°F); (6) provides valve actuation signals and monitors valve response; (7) monitors temperature of gases during spacecraft tank loading; and (8) measures flow rate of gas through reaction-control nozzles.

Test set is similar to that provided for Lunar Orbiter with the filter manifold assembly design the same as JPL drawing 8200740, "Fill Manifold Assembly."

4.3.1.5 Thrust-Vector-Control Servicing Unit

The thrust-vector-control servicing unit is mounted on a three-wheel trailer. The TVC servicing unit consists of: (1) A freon supply circuit; (2) a nitrogen pressure and purge circuit; and (3) a freon weighing section. A schematic of the unit is shown in Figure 4.3-4. The unit services the TVC system of the orbit-insertion motor and also contains the necessary control valves and monitoring devices for purging the Freon 114B2 tank prior to servicing. The unit delivers 437 pounds of sterilized freon to the spacecraft. Freon transfer is accomplished by pressurized nitrogen. The quantity of freon transferred to the spacecraft is accurately weighed by direct-reading platform scales. The unit is functionally similar to the HiBEX TVC servicing unit developed by Boeing.

4.3.2 Interfaces

Unless otherwise noted, the interfaces identified in the following are mechanical-type connections. Interface loads at these connections are either liquid or gas pressures (see Figure 4.3-5).

The unit interfaces with the spacecraft propulsion system as follows:

1) Propellant tankage--N_2 pressure to 315 psia;
2) N_2 tank--N_2 pressure to 3500 psia;
3) N_2 system plumbing--N_2 pressure up to 3500 psia;
4) Propellant tank pressure transducer--electrical;
5) Propellant system plumbing--N_2 pressure up to 315 psia;
6) Electrical solenoids--38 to 100 volts dc;
7) Midcourse propulsion engine jet vane--actuators;
8) Orbit-insertion TVC secondary injection control valves.

The unit interfaces with the spacecraft reaction control system:

1) Plumbing--N_2 pressure to 3500 psia;
2) Nitrogen tank for thrusting nozzles--N_2 pressure to 3500 psia;
3) Solenoids--Electrical 38 to 100 volts dc;

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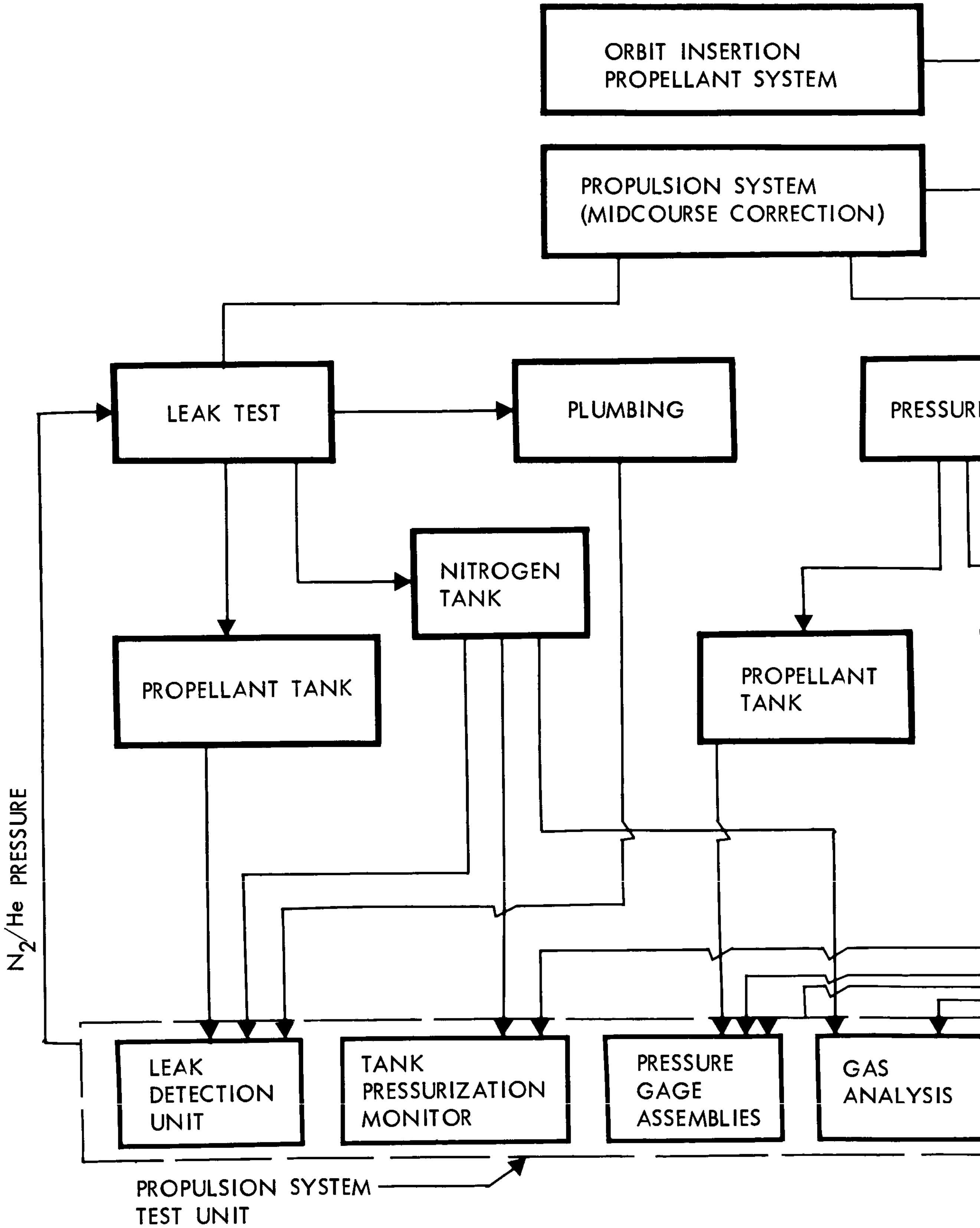
ORBIT INSERTION PROPELLANT SYSTEM
PROPULSION SYSTEM (MIDCOURSE CORRECTION)
LEAK TEST
PLUMBING
NITROGEN TANK
PROPELLANT TANK
PROPELLANT TANK
N_2/He PRESSURE
LEAK DETECTION UNIT
TANK PRESSURIZATION MONITOR
PRESSURE GAGE ASSEMBLIES
GAS ANALYSIS
PROPULSION SYSTEM TEST UNIT

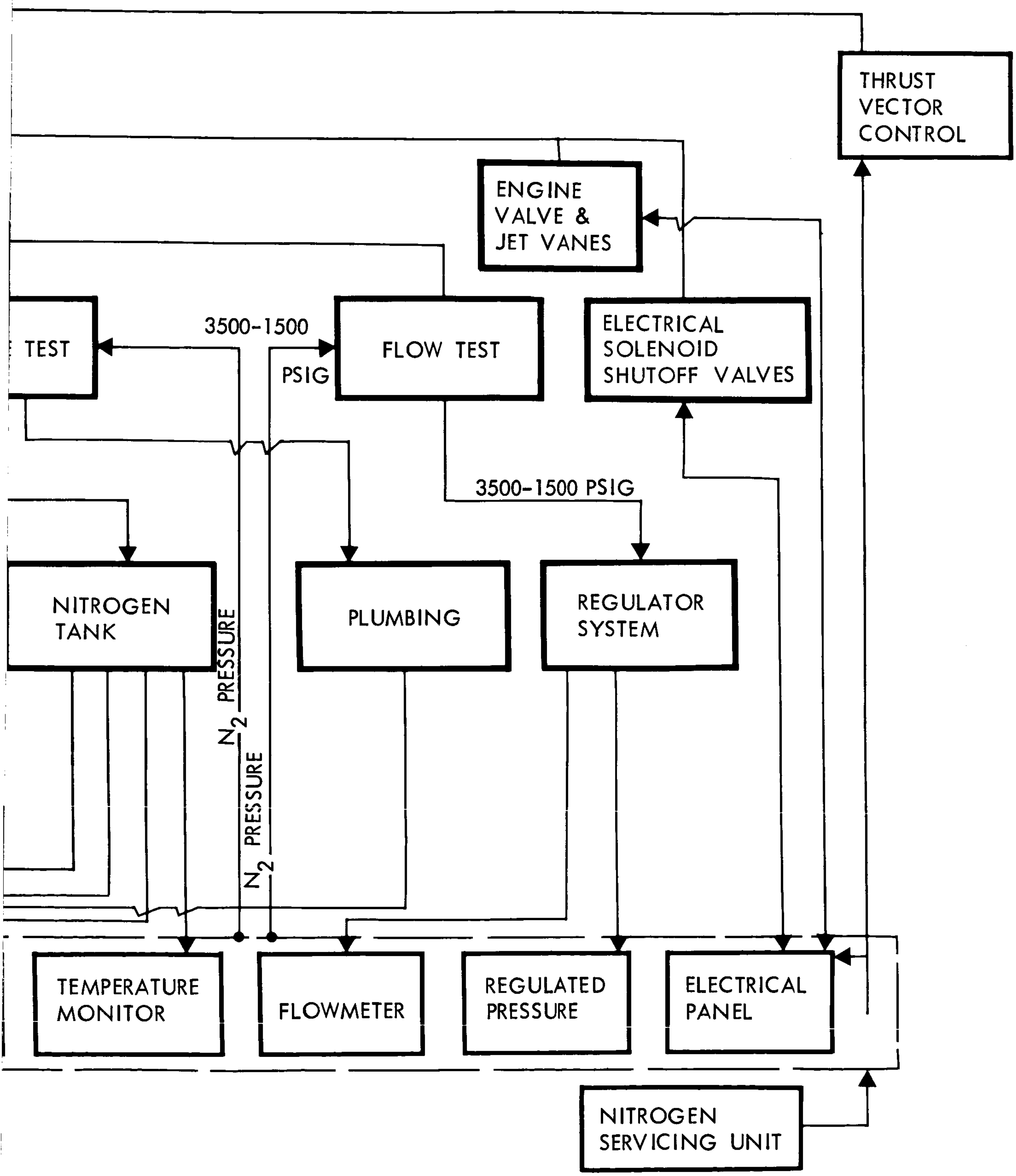

Figure 4.3-5: Propulsion Subsystem Interfaces

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The unit also interfaces with the nitrogen servicing unit and the electrical power supply.

4.3.3 Physical Characteristics

The major physical characteristics of this OSE are as follows:

1) All tanks and piping conform to the applicable ASME codes.
2) Gas pressure gages are equipped with blowout back panels or equivalent for personnel safety.
3) Equipment is designed such that a single failure will cause minimum impairment to system operation and hazard to personnel. Included in such preventive measures are relief and burst diaphragms for overpressures.
4) Manually operated controls and instrumentation panels are mounted with protection from inadvertent operation where such operation is critical.
5) The servicing and test units are wheel mounted.
6) The units are transportable by air, van, or truck.
7) The units are designed for ease of cleaning as required for use in clean-room areas. Smooth surfaces and tubular structure are employed and reentrant sections are avoided.

4.4 SCIENCE SUBSYSTEM OSE

Although it is presently planned that the Science Subsystem OSE be supplied as GFE, its key role in the success of the Voyager missions demands an understanding of its characteristics and functions by the spacecraft system contractor.

The purpose of the Science Subsystem OSE is to prove functioning of each experiment of the Science Subsystem in the environment in which the Science Subsystem operates.

The Science Subsystem OSE is used for subsystem testing through mission tests.

4.4.1 Applicable Documents

1) EDP 250, "Mariner Mars 1969 Orbiter Technical Feasibility Study," JPL, 16 November 1963.
2) OSE/MC-4-210A, "Functional Specification Mariner C, Operational Support Equipment Science Subsystem," JPL, 2 October 1964.
3) "Space Probes and Planetary Exploration," William R. Corliss, D. VanNostrand Company, Inc., Princeton, New Jersey, 1965.

4.4.2 Functional Description

The tentative concept concerning the Science Subsystem OSE follows.

4.4.2.1 General

1) Each sensor is mated with its electronics (i.e., if a failure occurs in either the sensor or its electronics, both will be replaced).
2) Each instrument has a flight calibration mode commanded either by the data automation equipment (DAE) or from the ground.
3) A history of the critical parameters of each instrument is available in a suitable form for insertion into the remote display memories.
4) At any time during the Voyager program, each Science Subsystem test set contains a record of the identity of each instrument in the subsystem. Techniques such as providing the test program and malfunction isolation program on magnetic tapes give the basic OSE the potential for growth that makes it useful for missions beyond 1971.

The Science Subsystem OSE consists of:

1) Science subsystem test set;
2) Spacecraft Bus simulator;
3) Science cable subsystem;
4) Alignment fixtures;
5) Noise checker;
6) Calibration standards;
7) Software.

The Science Subsystem test set block diagram (Figure 4.4-1) shows the major elements of the test set.

This test set is an extrapolation from the test set described in OSE/MC-4-210A. Two new features are added: a computer and a remote display. The three-fold increase in experiments over Mariner C, and the necessity to process the data in real time during system testing, indicate the need for a computer and remote display.

The computer contains the test program for primary checkout, and the malfunction isolation program for secondary tests. It processes the raw data from the test connectors into suitable form for the signal comparator. It also translates the raw data into engineering units for printout and display at the monitoring console and, for convenience of experimenters, at remote displays. Data from each instrument can be displayed. The experimenter can display previous history of the instrument simultaneously with the current output. Voice communications are provided for system-level testing.

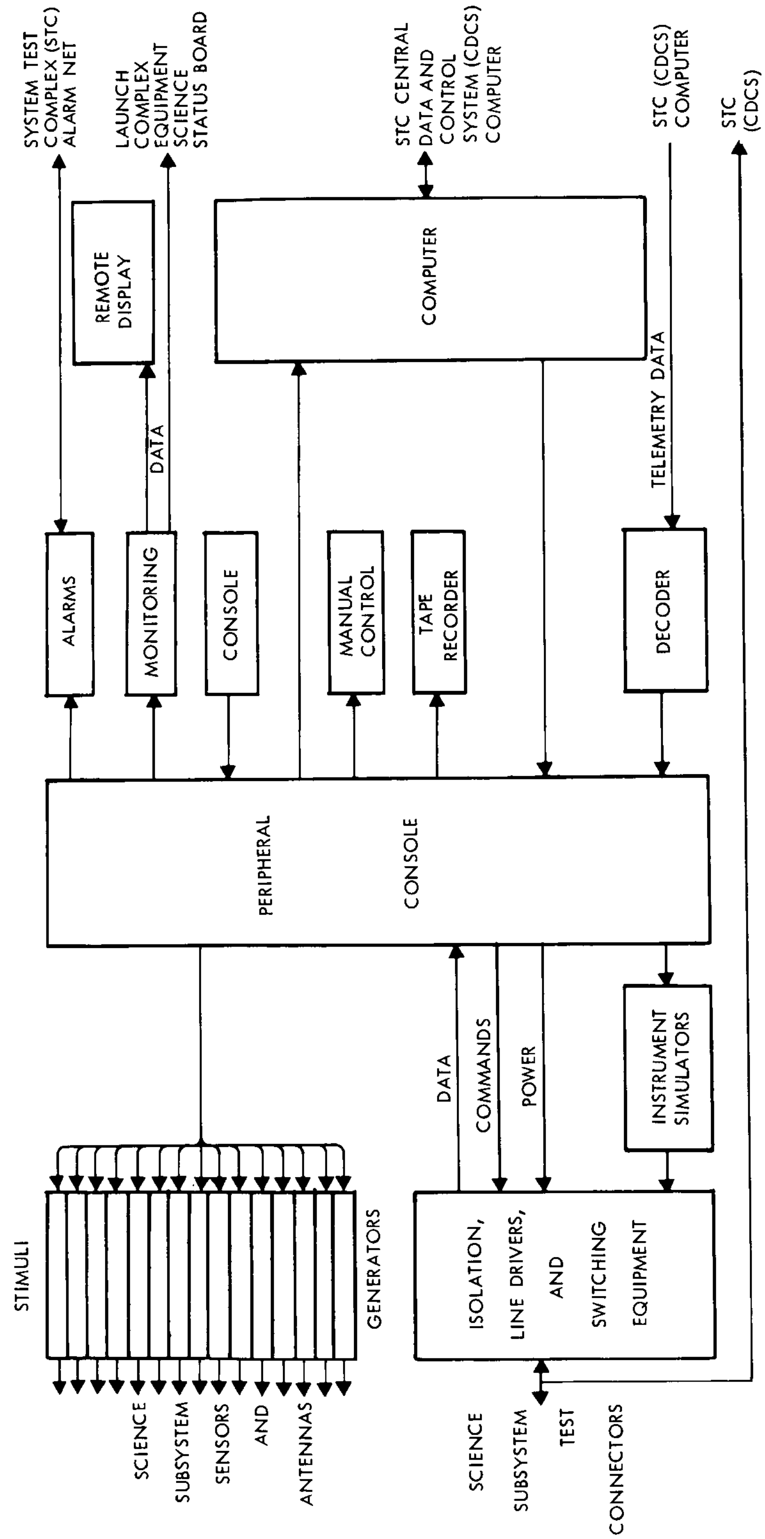


Figure 4.4-1: Science Subsystem Test Set

The Spacecraft Bus simulator provides mounting surfaces for the units of the Science Subsystem and the alignment fixtures in the absence of a Spacecraft Bus. It also supports the science cable subsystem. The bus simulator is used only in subsystem testing.

The science cable subsystem provides interconnection of the Science Subsystem and permits testing of the Science Subsystem in the absence of a spacecraft. The science cable subsystem is mounted on the Spacecraft Bus simulator.

The alignment fixtures provide a means of checking the alignment of the sensors and provide mounting surfaces for the stimuli generators.

The noise checker is used during subsystem integration to test the integrity of that portion of the cabling subsystem concerned with the Science Subsystem. It will provide nominal current flow through the power leads and will sample each signal lead.

Calibration standards are either secondary or tertiary and are used to test and calibrate the stimuli generators. Typical of these standards are photometer, rf, and time standards; ultraviolet (UV) and infrared (IR) sources; and radiation counters.

Handbooks are available covering the components of the Science Subsystem and the OSE. The handbooks for the Science Subsystem OSE describe operation, self-test procedures, and maintenance, and include a parts list.

A computer test program for the Science Subsystem is recorded on magnetic tape. The computer, following the instructions on the tape, applies the stimuli, selects signal paths, inserts nominal and tolerance information into the signal comparators, and reports the results of the test to the monitoring console and printout. In the case of a "no-go" result, malfunction isolation instructions contained in the test program tape are displayed at the monitoring console.

Upon successful completion of each level of test, a report is filed. The report includes:

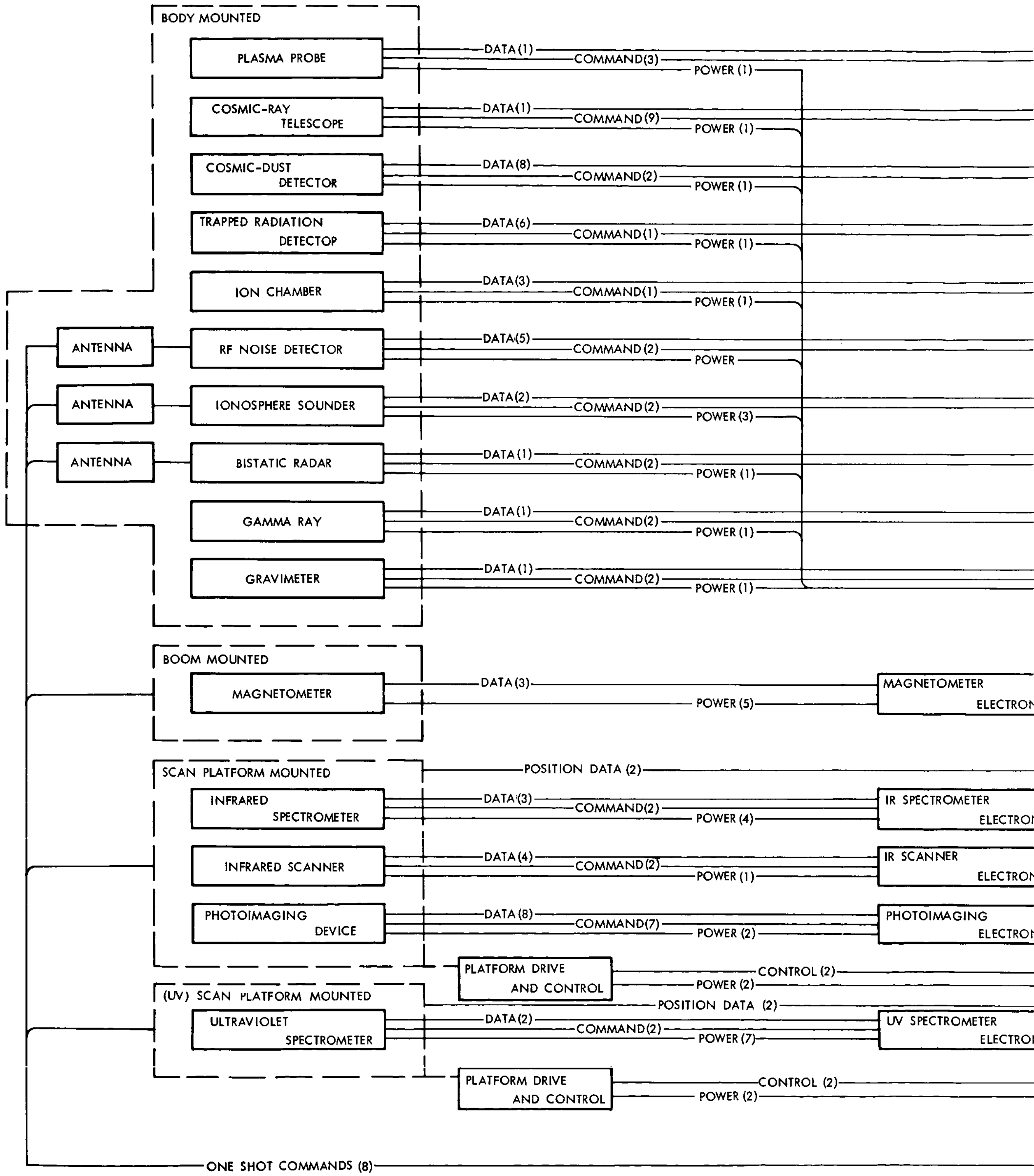
1) Level of test;
2) Time, date, and place of test;
3) Names and signatures of witnesses to the test;
4) A copy of the printout of the test;
5) A detailed account of malfunctions detected and methods used;
6) A detailed account of corrective actions taken.

Forms for these reports are provided.

4.4.2.2 Operation

The Science Subsystem is made up of 15 unique instruments and four pieces of auxiliary equipment. Figure 4.4-2 shows a block diagram of the subsystem. A typical series of tests involves:

1) Subsystem tests;
2) Subsystem verification tests;
3) Subsystem integration tests;
4) Seattle system tests;
5) KSC system tests;
6) Countdown and launch tests;
7) Mission tests.


BODY MOUNTED
PLASMA PROBE
DATA (1)
COMMAND(3)
POWER (1)
COSMIC-RAY TELESCOPE
DATA(1)
COMMAND(9)
POWER (1)
COSMIC-DUST DETECTOR
DATA(8)
COMMAND(2)
POWER (1)
TRAPPED RADIATION DETECTOP
DATA(6)
COMMAND(1)
POWER (1)
ION CHAMBER
DATA(3)
COMMAND(1)
POWER (1)
ANTENNA
RF NOISE DETECTOR
DATA(5)
COMMAND (2)
POWER
ANTENNA
IONOSPHERE SOUNDER
DATA(2)
COMMAND(2)
POWER (3)
ANTENNA
BISTATIC RADAR
DATA (1)
COMMAND(2)
POWER (1)
GAMMA RAY
DATA(1)
COMMAND(2)
POWER (1)
GRAVIMETER
DATA (1)
COMMAND(2)
POWER (1)
BOOM MOUNTED
MAGNETOMETER
DATA (3)
POWER (5)
MAGNETOMETER ELECTRON
SCAN PLATFORM MOUNTED
POSITION DATA (2)
INFRARED SPECTROMETER
DATA (3)
COMMAND(2)
POWER (4)
IR SPECTROMETER ELECTRO
INFRARED SCANNER
DATA (4)
COMMAND (2)
POWER (1)
IR SCANNER ELECTRO
PHOTOIMAGING DEVICE
DATA (8)
COMMAND(7)
POWER (2)
PHOTOIMAGING ELECTRO
PLATFORM DRIVE AND CONTROL
CONTROL (2)
POWER (2)
(UV) SCAN PLATFORM MOUNTED
POSITION DATA (2)
ULTRAVIOLET SPECTROMETER
DATA(2)
COMMAND(2)
POWER (7)
UV SPECTROMETER ELECTRO
PLATFORM DRIVE AND CONTROL
CONTROL (2)
POWER (2)
ONE SHOT COMMANDS (8)

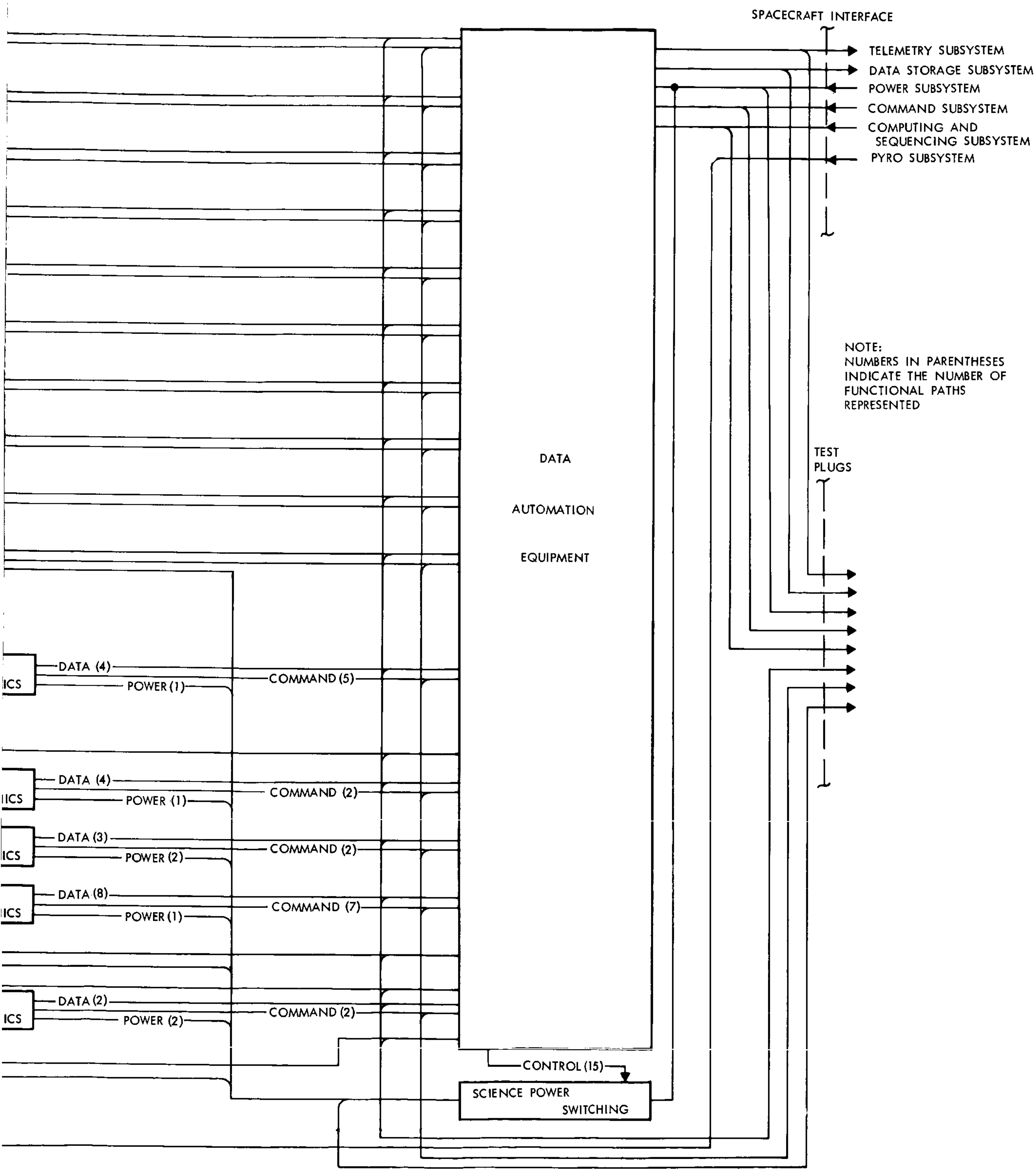


Figure 4.4-2: Typical Science Subsystem

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Subsystem Test--Scientific instruments are developed and tested individually. Upon receipt at JPL, these instruments are assembled and collectively tested as a subsystem, using appropriate stimuli. The individual parts of the subsystem are mounted on the Spacecraft Bus simulator. Mechanical interfaces and alignments are checked prior to electrical connection. Interconnection of the Science Subsystem is accomplished using the science cabling subsystem. The Science Subsystem test set is connected through test points provided. The test set provides power, commands, and stimuli to the subsystem in the primary test mode.

The test set receives data from test points located at the output of the DAE. The test set, when used for malfunction isolation, can read the outputs of the individual instruments or simulate these outputs and insert them into the DAE. The basic output of the test set is go/no-go information; however, a display of data is available at the monitoring console and the remote displays.

Subsystem verification tests identical to the subsystem tests by JPL are performed in the Seattle area.

Subsystem integration tests are performed with the spacecraft and the Science Subsystem simulator prior to integration of the Science Subsystem. In addition, a noise test is performed on the signal leads of the cable subsystem with nominal currents flowing through the power leads.

The Science Subsystem is then mounted on the Spacecraft Bus and mechanical interfaces and alignments are checked. The subsystem test is repeated prior to electrical interconnection with the remainder of the spacecraft. The Science Subsystem is then connected to the remainder of the spacecraft and the Science Subsystem test set is interconnected with the other subsystem OSE to form the System Test Complex (STC). A system test is then performed as described in the next section.

Seattle System Test--A system test using the STC is run a number of times (including the science quiet test) in Seattle and Eastern Test Range (ETR). In the Science Subsystem, the signal flow in a typical test mode is from the stimuli generator, through the Science Subsystem, to the telemetry and data storage subsystems. The signals are then returned to the STC central data and control (CDCS) via the telemetry stream. The CDCS routes the telemetry stream to the Science Subsystem OSE decoder. It then enters the Science Subsystem OSE peripheral console and is distributed in the same manner as in the subsystem test. Control of the stimuli generators is exercised by the STC through the Science Subsystem OSE computer. Power is supplied to the Science Subsystem from the spacecraft. During malfunction isolation, connections are made to the Science Subsystem through the test connectors and control of malfunction isolation is assumed by the subsystem OSE.

Countdown and Launch--Tests at this level are performed by the Launch Complex Equipment (LCE), which contains two STC's and, therefore, two sets of Science Subsystem OSE. Since the spacecraft is sealed in a nose fairing, the Science Subsystem OSE cannot apply the stimuli directly to the sensors. The function of the SS OSE is to test the calibration modes of the instruments. The Science Subsystem is commanded into a calibrate sequence. The SS OSE receives these commands and the telemetry stream th.. ough its decoder. Alterations of test sequence may be requested through a voice channel to the test director. The monitoring console provides the blockhouse with status (go/no-go) information about the subsystem. The remote display also monitors the telemetry stream.

Mission Tests--Tests during the mission are accomplished through the calibrate mode of each instrument. The calibrate mode is initiated either by sequencers on board the spacecraft, or by ground command through the Deep Space Network (DSN). No additional hardware at either the SFOF or DSIF's is required. The test results are transmitted from the spacecraft to a DSIF, through the GCS to the SFOF. The instrument records, developed during ground testing of the Science Subsystem, are removed and transmitted to the SFOF. The history of each instrument is available at the SFOF for use in calibration of the instrument and is an aid in interpreting the results of an experiment.

4.4.3 Interface Definition

The Science Subsystem OSE interfaces with these areas:

1) System Test Complex (STC);
2) Launch Complex Equipment (LCE);
3) Space Flight Operations Facility (SFOF).

4.4.3.1 STC Interface

There are five STC interfaces.

1) Power--The STC supplies power to operate the Science Subsystem OSE.
2) Telemetry Stream--The STC CDCS supplies the telemetry stream to the Science Subsystem decoder.

3) Subsystem Test Information--The Science Subsystem OSE supplies the STC CDCS with raw data from the subsystem test connectors.

4) Alarm--This is a two-way interface between the test set alarm and the STC alarm net.

5) Computer--During system testing, control of the Science Subsystem OSE Computer is assumed by the STC CDCS computer.

4.4.3.2 LCE Interface

During countdown and launch, the test set monitoring console provides status (go/no-go) information to the blockhouse Science Subsystem status board.

4.4.3.3 SFOF Interface

There are no hardware interfaces, but the instrument records must be compatible with SFOF readout equipment.

4.4.4 Physical Characteristics

All components of the Science Subsystem OSE, with the exception of the stimuli generators, are mounted on 19-inch-wide panels and enclosed in standard JPL equipment consoles. The monitoring console, alarms, and manual controls are grouped together (see STC section for drawings). The isolation and switching equipment is mounted as close to the spacecraft as possible. The stimuli generators are mounted on the alignment fixtures while in use, and are stored in two 36-by 72-by 18-inch cabinets. The remote displays consist of four standard JPL equipment racks, located in a separate area near the Science Subsystem OSE.

4.4.5 Reliability and Safety

The design of the Science Subsystem OSE is based on the conservative use of proven components, and the status of the OSE is monitored periodically by the use of a self-check routine. In addition, self-checks are interleaved between tests. Typically, the input-output paths are set up, nominal and tolerance information is set into the comparators, commands are generated, and then a self-check path is closed and the commands verified. A signal is directed through the input self-check and output paths into the comparator and, if all is normal, an enable signal is generated to start the test. In addition, test connectors are mechanically keyed and electrically interlocked to avoid mismating. All power to the subsystem is monitored continually and a dangerous condition causes an alarm and shuts down both the OSE and the subsystem. Interfaces with the STC and LCE are also interlocked and monitored. With these precautions, the safety of the subsystem, STC, and LCE is assured.